STACKS

JET PROPULSION

Journal of the
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Rocketry . . . . . Jet Propulsion Sciences . . . . . Astronautics

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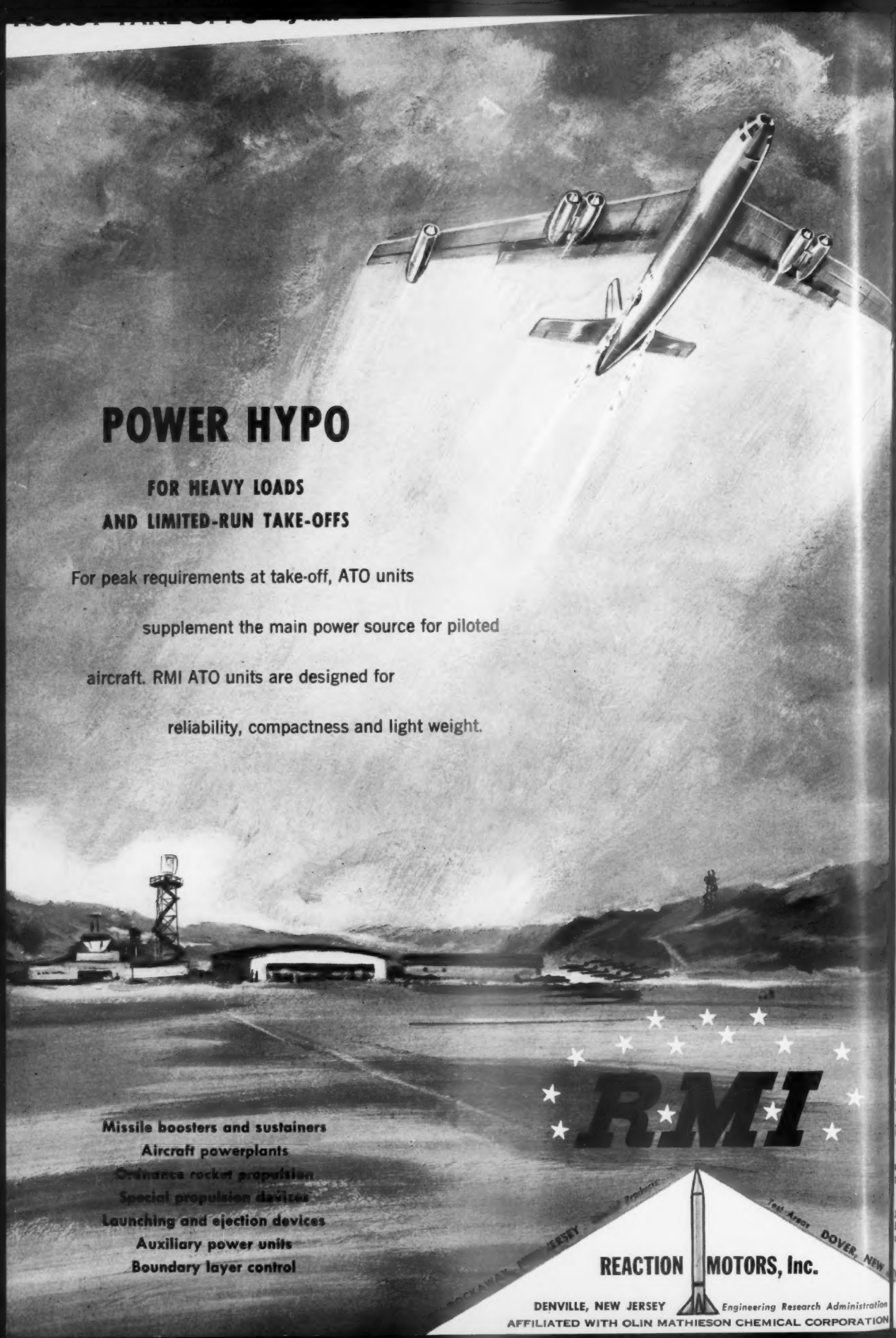
SCIENCE AND INDUSTRY

The Mechanics of Film Cooling—Part 1 . . . . . *Eldon Knuth* 359
Transpiration Cooling in Porous Metal Walls . . . *E. Mayer and J. G. Bartas* 366
A New Supply System for Satellite Orbits—Part 2 . . . *Krafft A. Ehricke* 369
Thermal Properties and Applications of High-Temperature Aircraft Insulation . . . . . *Paul Greebler* 374
Index of Papers Published in Volume 24, 1954 387

Technical Notes 379
Kretschmar, on Thermal Expansion of Liquid Propellants
Ehricke, on Optimum Transfer Orbits in Space
Manildi, on Response Time Lag of Temperature Instruments
Stehling and Foster, on Liquid Jet Atomization
Gongwer, on Duct Losses in Jet Engines
Glassman and Scott, on Ethylene Oxide Monopropellant
Jet Propulsion News 389
ARS News 396
Book Reviews 401
Technical Literature Digest 405

ARS ANNUAL CONVENTION N.Y.C., NOV. 30-DEC. 1 PROGRAM ON P. 396

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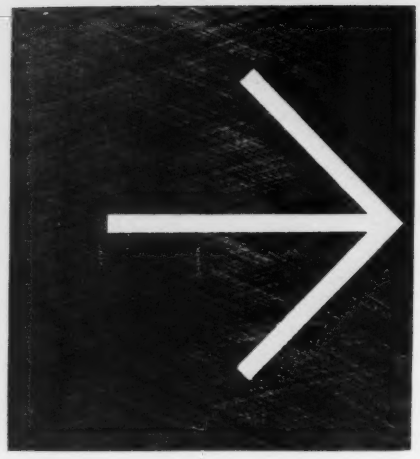
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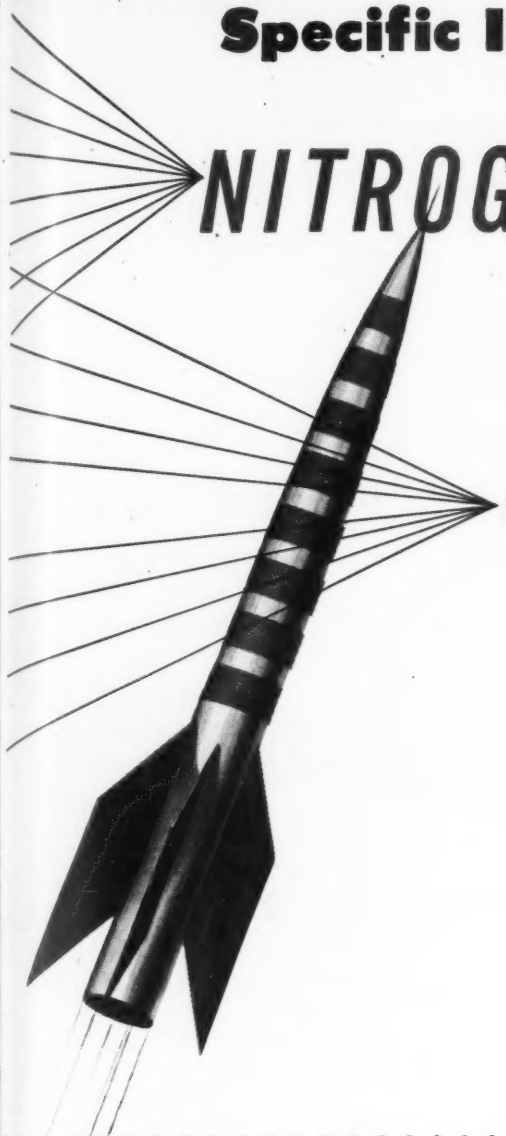
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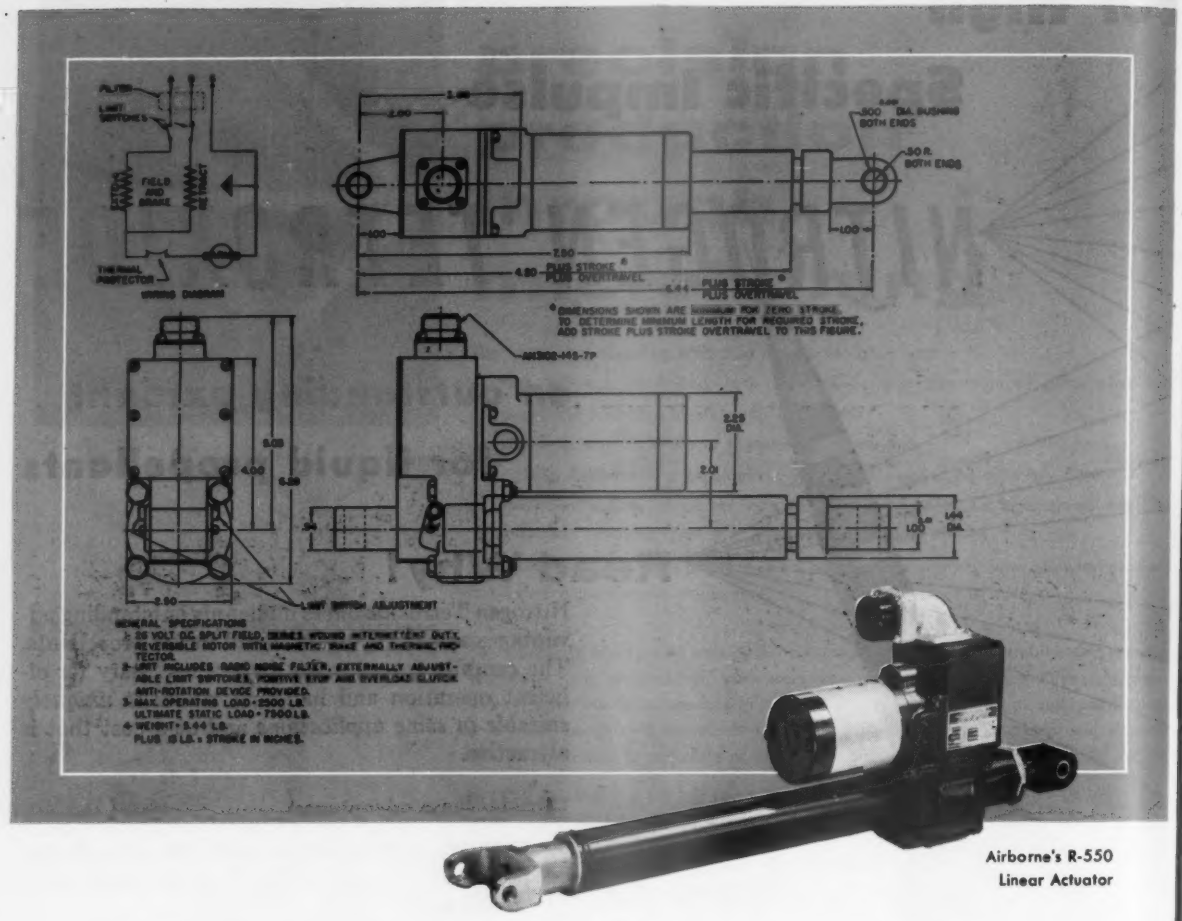
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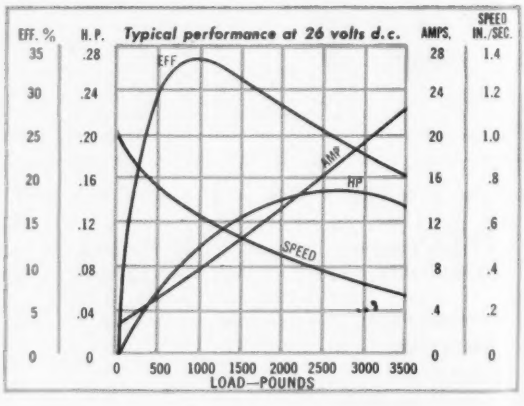
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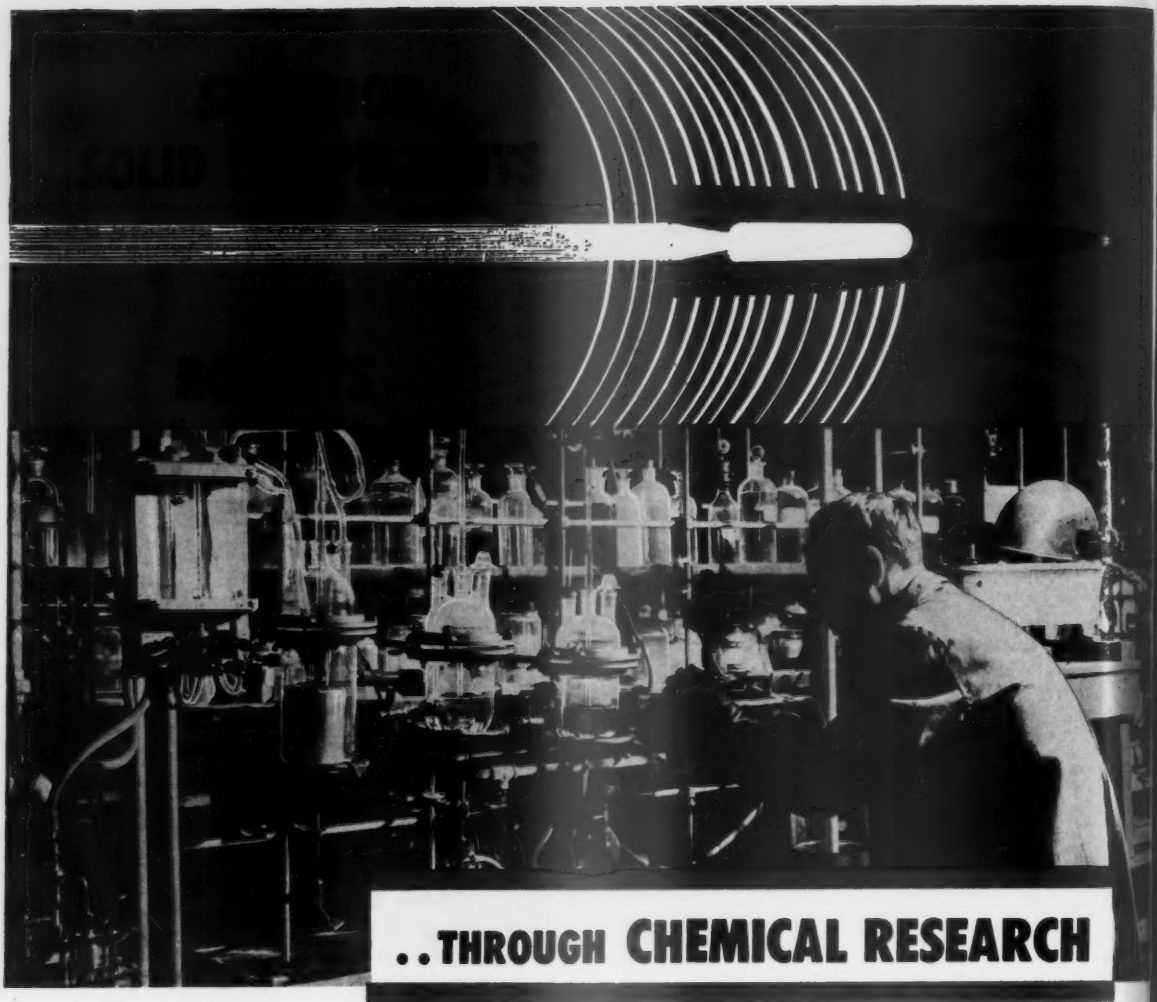
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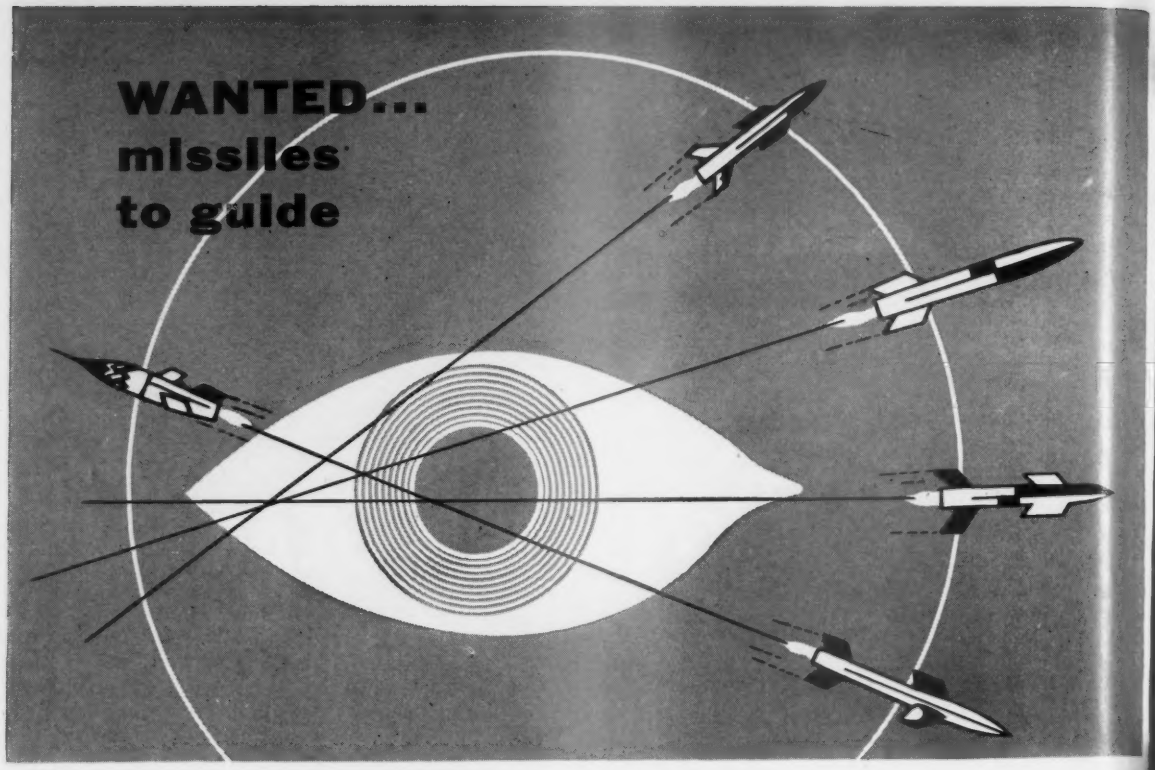
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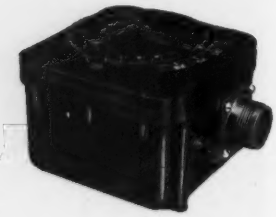


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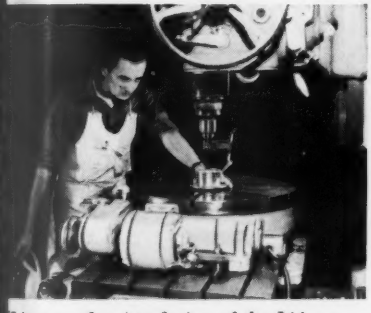
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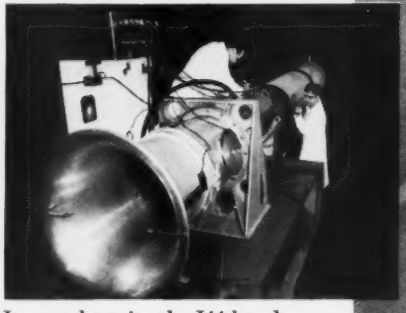
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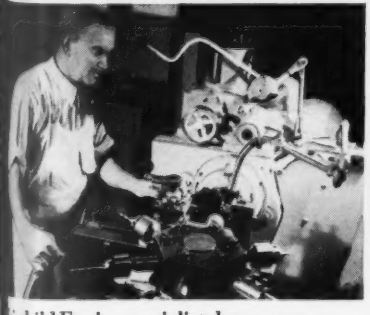
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Starting with the January 1955 issue, the Journal of the American Rocket Society will appear on a monthly basis

The Mechanics of Film Cooling[1]—Part 1

ELDON L. KNUTH[2]

Aerophysics Development Corporation, Pacific Palisades, Calif.

Thin liquid wall films flowing under the influence of high-velocity turbulent gas streams were studied for the purpose of obtaining an understanding of the mechanics of film cooling. Conditions which insure liquid-film attachment to solid surfaces without loss of unevaporated liquid to the gas stream when simple radial-hole injectors are used were found; the maximum allowable coolant-flow rate for a stable coolant film was determined (a stable coolant film is obtained when no unevaporated coolant is entrained by the gas stream as the result of interfacial disturbances); and a method for calculating the evaporation rate and the surface temperature for a stable inert coolant film was found.

Nomenclature

a = velocity of sound
c_p = specific heat at constant pressure
C_a = modified cavitation parameter
C_f = gas-stream friction coefficient
C_h = gas-stream, heat-transfer coefficient
C_m = gas-stream, mass-transfer coefficient
d = duct diameter
D = molecular mass diffusivity
E = evaporation coefficient
H = enthalpy
ΔH = coolant heat of vaporization
k = thermal conductivity
L = liquid-film or test-section length
m = mass transfer per unit area and per unit time
p = pressure
Pr = Prandtl number
q = heat transfer per unit area and per unit time by conduction
r = distance from the center line of the pipe
R = gas constant
Re = Reynolds number
Sc = Schmidt number
T = temperature
u = velocity in the x-direction
u^* = $u/\sqrt{\tau_0/\rho_0}$
v = velocity in the y-direction
v_L = liquid velocity averaged over cross-sectional area of injection orifices
w = oxidizer transfer per unit area and per unit time
y = distance into gas stream from gas-stream bounding surface measured perpendicularly to bounding surface
y^* = $\rho_0\sqrt{\tau_0/\rho_0}\, y/\mu_0$
Γ = liquid flow per unit time and per unit length of tube circumference
δ = gas-stream laminar sublayer thickness
δ' = $\int_0^\delta (\mu_{M0}/\mu_M)\, dy$
δ^* = $\rho_\infty\sqrt{\tau_\delta/\rho_\infty}\int_0^\delta (1/\mu_M)\, dy$
η = film thickness averaged with respect to x and t
η^* = $\rho_{Lf}\sqrt{\tau_0/\rho_{Lf}}\eta/\mu_{Lf} \approx \sqrt{2\Gamma/\mu_{Lf}}$
θ = oxidizer specific concentration (weight of oxidizer per unit total weight)
μ = dynamic viscosity
π = 3.14
ρ = density
τ = shearing stress

Subscripts

G = gas
Lf = liquid in the film
Li = liquid in the injector orifice
M = mixture of gas and vapor
0 = bounding surface of gas stream (a liquid film surface or a duct wall)
s = saturation conditions corresponding to T_0
t = total
V = vapor
δ = junction of turbulent core and laminar sublayer in gas stream
∞ = bulk property or average velocity

1 Introduction

FILM cooling[3] is the protection of a given surface from injurious effects of a proximate heated fluid stream by the interposing of a thin continuous protective liquid film between the given surface and the fluid stream. Its use is justified when the proximate fluid stream is extremely hot and when a more satisfactory method for protecting the given surface from destruction by heat is not available; or when the proximate fluid stream reacts chemically with the given surface, when such chemical reaction is undesirable, and when a more satis-

Received February 22, 1954.

[1] This paper is based on part of the thesis submitted by the author in partial fulfillment of the requirements for the degree of Doctor of Philosophy in Aeronautical Engineering, California Institute of Technology, and presents the results of one phase of research carried out at the Jet Propulsion Laboratory, California Institute of Technology, under Contract No. DA-04-495-Ord 18, sponsored by the Department of the Army, Ordnance Corps.

[2] Propulsion Engineer.

[3] Although the flow of the thin liquid wall films under the influence of high-velocity turbulent gas streams and/or the unidirectional turbulent diffusion of one gas through another gas occurs in numerous engineering applications (e.g., in evaporators, condensers, two-phase combustion processes, and film-cooling systems), attention will be concentrated, for the sake of convenience, on that application which motivated the present study—film-cooling systems.

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factory method for separating the injurious fluid stream from the given surface is not available; or when the proximate fluid stream carries with it materials which are deposited easily on solid surfaces and when deposits of this nature are undesirable on the given surface. Structural members which are likely to be film-cooled include combustion chambers,[4] where the heated fluids are the products of combustion, and external surfaces of high-velocity missiles, where the heated fluids are aerodynamically heated atmospheric gases. Inability to cool these structural members adequately could limit seriously the performance of the corresponding machines.

The benefits of film cooling are realized most fully when the liquid film is attached to the surface without loss of unevaporated liquid to the proximate fluid stream, when the attached liquid film is stable (i.e., no unevaporated liquid is lost to the proximate fluid stream as the result of disturbances on the film surface), and when the evaporation rate is low.[5] It is important, therefore, to know how these three conditions may be obtained. Little basic information was available in 1950, however, regarding the prerequisites for these conditions. Consequently, a study was initiated during that year for the purpose of obtaining a basic understanding of thin liquid wall films flowing under the influence of high-velocity turbulent gas streams by investigating the attachment of liquid films to solid surfaces in the presence of high-velocity gas streams, the conditions which are sufficient for stability of liquid wall films flowing under the influence of high-velocity turbulent gas streams, and the rate of evaporation from a stable liquid wall film into a heated turbulent gas stream. The results of that study (2)[6] are summarized in this paper.

2 Experimental Equipment

A Equipment for Film-Attachment Studies

The flow diagram for the equipment used in the film-attachment studies (4) is presented in Fig. 1. The equipment consisted essentially of an air-supply system, a liquid-supply system, a Lucite test section, and appropriate controlling and measuring devices.

The air was obtained from the standard air-supply system in general use at the Jet Propulsion Laboratory; its flow rate was measured with the aid of a sharp-edge orifice which had been designed according to ISA specifications, a 30-in. mercury manometer, a Bourdon-type pressure gage, and an iron-constantan thermocouple. After being passed through the measuring orifice, the air was conducted through the Lucite test section (located 46 diameters downstream from the measuring orifice) and exhausted to the atmosphere through a manually operated gate valve (located 16 diameters downstream from the test section). The magnitude of the static air pressure in the test section, which was controlled by the gate valve, was measured with a Bourdon-type pressure gage, and the air temperature in the test section was measured with the aid of an iron-constantan thermocouple.

The liquid used in the tests was stored in a metal tank pressurized by nitrogen gas from a commercial compressed-nitrogen bottle. From the storage tank the liquid was conducted through a rotameter, a manually operated needle valve, a screen filter, and finally into the test section through one of the four available injector openings. The liquid temperature was measured at a point several inches upstream from the injector with the aid of an iron-constantan thermocouple; electric potentials from the thermocouples were measured with the aid of a hand-balanced potentiometer.

The Lucite test section was fabricated from commercially available stock having a 3-in. inside diameter and a $^1/_4$-in. wall thickness. Lucite was selected because of its transparency and good machining properties. Four radial holes, ranging from $^1/_{16}$ to $^5/_{32}$ in. in diameter, were drilled at 90-deg intervals around the periphery of the test section. During any given test, liquid was injected through only one of these holes; the other holes were utilized as static-pressure taps and thermocouple sites.

A total-pressure probe attached to a manually operated micrometer screw was used for the purpose of investigating the velocity profile of the air flowing in the test section. Tests were conducted only after the velocity profile was found to be symmetric with respect to the center line of the duct.

B Equipment for Film-Stability and Evaporation-Rate Studies

The flow diagram for the equipment used in the film-stability and evaporation-rate studies is presented in Fig. 2. The equipment consisted essentially of a gas-supply system, a liquid-supply system, several interchangeable test sections, and appropriate controlling and measuring devices.

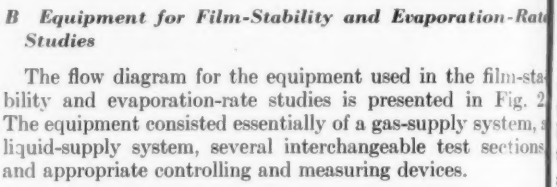


Fig. 2 Flow diagram for equipment used in research on stability of and evaporation from liquid wall films

The gas was in general produced by burning fuel (Union Oil Company's No. 1 thinner) and air in a modified turbojet combustion can. The fuel flowed into the burner from a nitrogen-pressurized storage tank; its flow rate was measured with the aid of Bourdon-type pressure gages and a calibrated injection nozzle. The air was obtained from the standard air-supply system in general use at the Jet Propulsion Laboratory; its flow rate was measured with the aid of a sharp-edge orifice which had been designed according to ISA specifications, a 50-in. mercury manometer, a Bourdon-type pressure gage, and a chromel-alumel thermocouple. The products of combustion were mixed and then calmed in an insulating settling chamber having a cross-sectional area eighteen times that of the approach duct. After passing through the settling chamber, the combustion products flowed first through

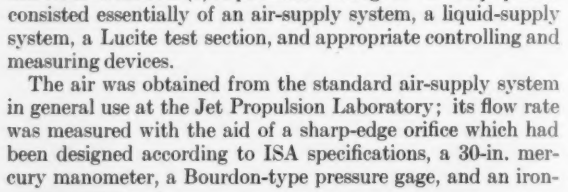


Fig. 1 Flow diagram for equipment used in research on attachment of liquid wall films

[4] Historial Note: It is believed that credit for the first application of film cooling is due Robert H. Goddard, pioneer of rocketry in America, who is known to have film-cooled the combustion-chamber walls of experimental rocket motors in the year 1929 (Ref. 1, p. 3).

[5] If the given surface is the inner wall of a rocket motor, it is desirable also that the liquid be reactive with the combustion products, thereby acting, at least to some extent, as an injected propellant as well as a film coolant; that the liquid be one of the propellants, so that no special supply system is required for the coolant; and that an inexpensive, noncritical, light-weight material (perhaps aluminum; see Ref. 3) be used in the construction of the combustion-chamber wall.

[6] Numbers in parentheses refer to References on page 365.

27 diameters of insulated, straight, constant-diameter approach duct and then into the test section. The total gas temperature in the settling chamber was measured with the aid of a shielded chromel-alumel thermocouple; the magnitude of the static pressure in the approach section immediately upstream from the injector was measured with a Bourdon-type pressure gage; and the pressure drop in the test section was measured with the aid of a 50-in. water manometer.

Water (procured directly from the Laboratory water supply pipe at approximately 100 psig) and aqueous sucrose solutions (produced by mixing tap water and reagent sucrose and stored in a nitrogen-pressurized supply tank) constituted the liquids in the film-stability tests; water was the only liquid

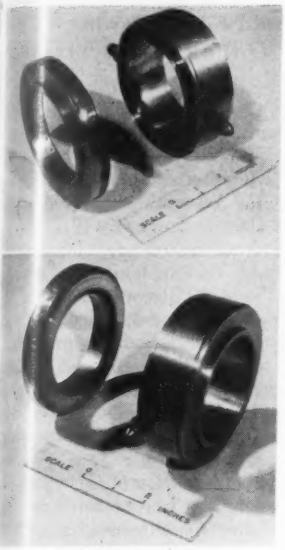


Fig. 3 Views of the coolant injector used in research on liquid wall films

used in the evaporation-rate tests. After leaving the Laboratory water-supply line (or the supply tank, as the case may be), the liquid flowed (in the sequence named) through one of two manually operated needle valves, one of two rotameters, a filter, the coolant-injector plenum chamber (Fig. 3), twenty-four (twelve, for one group of tests) 0.011-in.-square passages (characterized by relatively high flow resistances and serving to nullify gravity effects in the plenum chamber), and corresponding equally spaced, $^1/_{16}$-in.-diameter, radial-injection holes. The liquid was attached to the inner wall of the test section by the action of the high-velocity gas stream. For tests employing unheated gases, the film temperature was assumed to be equal to the wet-bulb temperature in the case of a water film and the temperature (measured) of the exhausted liquid in the case of a film of aqueous sucrose solution; for tests employing heated gases, the film temperature was calculated using Equation [39] of this paper.

For use in tests employing heated gases, a test section was fabricated from a 5-ft length of 347 stainless-steel tubing having a 0.063-in. wall and a 3-in. outside diameter. The inside diameter of the tube was honed to 2.90 in. (leaving a wall thickness of 0.050 in.), and 120 thermocouples were spot-welded to the outside of the tube, one every inch (measured axially) at the 12 o'clock position and one every 3 in. at the 3, 6, and 9 o'clock positions. For use in tests employing gases at ambient temperature, several lengths of Lucite tubing having a 3-in. inside diameter and a 0.25-in. wall were procured.

The electric potentials from the thermocouples which were used in order to facilitate the measurement of the gas temperature and eight of the test-section-wall temperatures were recorded on a 12-point, 50-mv-range Brown recorder located in the control room. The remaining 112 thermocouples were connected in turn to a continuously recording, single-channel potentiometer located at the central recording room of the Jet Propulsion Laboratory by means of a 25-point, 10-level, telephone-type automatic electric stepping switch capable of scanning the 112 thermocouples in less than 2 min.

3 Liquid-Film Attachment With Radial Injector Holes and a High-Velocity Gas Stream

A Previous Studies

Previous systematic studies of the liquid-film-attachment problem were limited to the experimental determination of the critical velocity of injection when a slot around the duct circumference is used (5, 6, 7), where the critical velocity of injection was defined as the maximum mean velocity of the liquid in the injection slot obtainable with no visible separation of the liquid film from the inner surface of the test section. The authors of these papers asserted that the use of injection slots would provide more uniform liquid films and more easily controlled liquid-flow rates than could be obtained with other film-attachment methods. Remarks on these assertions will be made in the following sections 3-C and 4-B.

B Experimental Results

An experimental study was conducted in order to determine the critical velocity of injection for attachment of a liquid film to the inside of a circular duct when radial injector holes and effects of a high-velocity gas stream are used (4). The critical velocity of injection was defined as the maximum mean velocity of the liquid in the injector orifices obtained with no visible separation of the coolant film from the inner surface of the test section.

Liquid viscosities from 1.59×10^{-5} to 10.69×10^{-5} lb sec/ft^2, liquid densities from 1.94 to 3.02 slugs/ft^3, and liquid vapor pressures from 0.15 to 3.85 psia were secured by using water, aqueous zinc chloride solutions, aqueous sucrose solutions, and carbon tetrachloride. Test-section air densities from 2.26 to 9.58×10^{-3} slug/ft^3 were provided by a gate valve at the discharge end of the air duct; no other appreciable gas-property variations were realized. Injector-orifice diameters varying from $^1/_{16}$ to $^5/_{32}$ in. were achieved by using different injectors; no gas-duct-diameter variations were realized.

For each test point, the air-supply system was operated at the desired output until a stable air-flow rate was obtained. When this condition was realized, the liquid-flow rate was increased slowly from the no-flow value to a value when the critical velocity of injection was reached. At that instant the air-flow rate, the injected-liquid-flow rate, and the appropriate temperatures and pressures were recorded. The data taken at the critical injection velocity are given in Table I of Ref. 4.

Incidental to the obtainment of attachment data, the circumferential spread of the liquid film corresponding to liquid flows at 21 different critical injection velocities was measured at a point located arbitrarily 1.25 in. downstream from the injection point. The spread appeared to be chiefly a function of the liquid-flow rate; i.e., the fluid properties, the air velocity, and the injector-hole diameter had apparently a minor effect on the spread for the range of fluid properties and air velocities tested. Typical values of the spread were 0.75 in. for 1.5×10^{-4} slug/sec and 1.50 in. for 6.0×10^{-4} slug/sec of liquid flow. Since the spread would vanish as the liquid-flow rate approached zero, the data would seem to indicate that the circumferential spread is proportional to the square root of the liquid-flow rate.

C Discussion of Data

Since no theoretical analysis of the film-attachment problem has been made, the data were plotted in dimensionless form (Fig. 4); the abscissa is a function of gas-stream Reynolds number Re_G, liquid-stream Reynolds number Re_{Li}, and

a modified cavitation parameter Ca_{Li}, and the ordinate is the ratio of the gas- and liquid-stream momenta $\rho_\infty u_\infty^2/\rho_{Li}V^2$. Here Re is Reynolds number based on diameter, Ca is a modified cavitation parameter defined by $Ca = \rho_{Li}V^2/(p - p_V)$, p is pressure, V is liquid velocity averaged over the cross-sectional area of the injection orifices, and the subscripts G, Li, and V refer to gas, liquid in the injector orifice, and vapor, respectively. The exponents of the dimensionless parameters were determined from crossplots on log-log paper of liquid- and gas-stream parameters for a constant value of the momentum ratio. (The line drawn in Fig. 4 has a scope of 0.8.) The di-

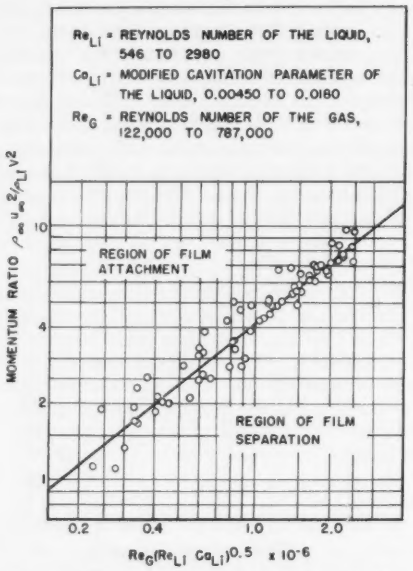


Fig. 4 Dimensionless plot of film-attachment data taken at critical velocity of injection

mensionless parameters used are speculative to the extent that the gas viscosity and test-section diameter were held constant and the vapor pressure of the liquid was varied over only a small range during the tests; e.g., with available data it was not possible to discern whether the significant gas-stream Reynolds number is that (based on gas velocity u_∞) which describes the gas stream in general or that (based on friction velocity $\sqrt{\tau_0/\rho_0}$) which describes only the gas flow near the duct wall.

It is concluded that effective film attachment may be obtained with simple radial-hole injectors in the presence of a high-velocity gas stream over a useful range of operating conditions so that the study of more complex means of injection (such as porous walls and circumferential slots) and of insuring film attachment probably is not required. It is concluded furthermore that the critical velocity of injection is a definite function of certain parameters of the system; thus it is possible to predict beforehand whether a given film-coolant stream will attach to the wall or will continue on into the gas stream.

Fig. 4 should be a useful guide in the design of film-cooling systems for turbulent flat-plate flows as well as for turbulent duct flows. Until further information is obtained, it is suggested that twice the local boundary-layer thickness be taken to be the characteristic gas-stream dimension in the case of flat-plate flows; this dimension corresponds to the duct diameter which was used in the case of pipe flows. Obviously, caution should be exercised when extending the presented results to parameter values outside the investigated range.

Since the calculation of streamlines for viscous fluid flows around sharp corners is an extremely difficult task, it seems unlikely that an analytical study of the film-attachment problem would be a rewarding undertaking at the present time. Experimental methods, on the other hand, have already been used successfully for the determination of the relative importance of several parameters involved in the film-attachment process. Therefore, if it should be deemed necessary to obtain further information concerning liquid-film attachment, it is suggested that (at the present time) experimental studies might be most profitable.

As mentioned in section 1, it is possible that film cooling will be used for (among other things) the protection of the external surfaces of guided missiles and the internal surfaces of rocket-motor nozzles. If simple film-coolant injector holes are to be located on these surfaces, it would be necessary to study the film-attachment phenomena at supersonic gas velocities and for various surface geometries.

4 Stability of Liquid Films Flowing Under the Influence of Turbulent Gas Streams

A Previous Studies

The stability of thin liquid wall films flowing under the influence of high-velocity turbulent gas streams has been investigated experimentally by Kinney and Abramson of the National Advisory Committee for Aeronautics (8) and to a limited extent by Greenberg of Purdue University (7).

Kinney and Abramson observed visually annular liquid flow with concurrent turbulent air flow in horizontal transparent tubes. The surface of the liquid film was observed to be relatively smooth at low liquid-flow rates and disturbed markedly at higher liquid-flow rates. The authors reported that the liquid flow per circumferential length at which marked liquid-flow disturbances initially occurred increased with increased liquid viscosity, increased slightly with decreased liquid surface tension, and did not vary appreciably with changes of air-mass velocity.

Kinney and Abramson presented a portion of their data in the form shown in Fig. 5, where the heat-transfer results from a report by Kinney and Sloop (9) are included as well as the research results reported in Ref. 8. The curve is the dimensionless velocity correlation for adiabatic single-phase pipe flows which was given by Deissler (10); the abscissa is the dimensionless distance y^*, defined by $y^* = \rho_0\sqrt{\tau_0/\rho_0}(y/\mu_0)$, and the ordinate is the dimensionless velocity u^*, defined by $u^* = u/\sqrt{\tau_0/\rho_0}$, where τ is shearing stress, ρ is density, y is distance into the fluid stream measured perpendicularly to the duct wall, μ is dynamic viscosity, u is axial velocity, and the subscript 0 refers to the duct wall. The break in the curve of Fig. 5 indicates approximately the transition from the laminar region to the turbulent region. The points which have been singled out on the curve correspond to the values of the dimensionless film thicknesses at which the liquid-film

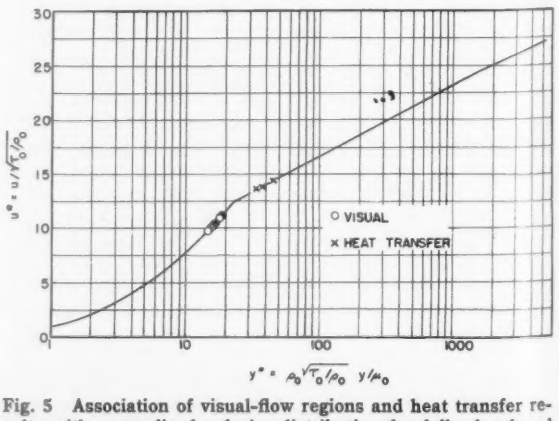


Fig. 5 Association of visual-flow regions and heat transfer results with generalized velocity distribution for fully developed flow in smooth tubes applied to liquid films (from Fig. 6 of Ref. 8)

surface initially became rough; they were singled out in order to indicate the magnitude of the critical dimensionless film thickness relative to the dimensionless laminar sublayer thickness for single-phase pipe flows. Kinney and Abramson subscribed to the hypothesis, suggested earlier by Colburn and Carpenter (11), that an annular liquid film flowing under the influence of a high-velocity turbulent gas stream behaves as the wall layer with the same thickness and shearing stress in single-phase liquid flow, and that it is essentially laminar when its thickness is less than the thickness of the laminar sublayer in the corresponding single-phase flow. Remarks on this hypothesis are included in section 4-C.

Of the quantitative results for flows in transparent tubes which were presented by Kinney and Abramson, it appears that those depending upon values of fluid properties which vary appreciably with temperature changes (especially those depending upon values of liquid viscosities) contain slight errors. The authors calculated fluid properties, assuming the film temperature to be the same as that of the entering fluids (i.e., 80 F); actually, the film temperature was more nearly the wet-bulb temperature corresponding to the relative humidity of the gas stream [see, e.g., Equation [39] of this paper]. Since the gas was virtually dry air (according to a personal communication from Kinney), this wet-bulb temperature was approximately 50 F, a temperature considerably lower than the 80 F of the entering fluids.

Greenberg, in his M.S. thesis, included nine photographs of annular liquid flow with concurrent air flow in a horizontal Lucite tube. In the first of these photographs (Fig. 23 of Ref. 7) the surface of the liquid film appeared to be relatively smooth. The other eight photographs, taken at higher liquid-flow rates, indicated varying degrees of liquid-film surface disturbance. No correlation of the inception points of the marked disturbances was attempted by Greenberg in the thesis.

B *Experimental Results*

During the present investigation, the dividing line between the areas of stable and unstable flow of annular liquid wall films in the presence of high velocity turbulent gas flow in a duct was detected indirectly, by inspection of mass-transfer data when heated gases were employed, and directly, by examination of the liquid film itself, when gases at room temperature were employed.

Data concerning the rate of mass transfer into a turbulent heated gas stream from a liquid wall film (averaged over the length of the film) were obtained for specified operating conditions by operating the test apparatus at the desired gas-flow rate, gas temperature, and liquid-flow rate until steady-state conditions were attained and then by recording (together with other pertinent data) the electric potentials created within thermocouples which were spot-welded to the exterior of the metal test duct. The axial position along the duct at which the wall temperature varied rapidly from a value below that of the boiling temperature of the liquid to a value approaching that of the hot gas stream was assumed to coincide with the axial position along the duct, corresponding to the end of the protective liquid film. Tests were conducted for gas-stream-diameter Reynolds numbers from 105,000 to 433,000, gas-stream temperature from 1103 to 2239 R, water-flow rates from 0.012 to 0.20 lb/sec/ft of tube circumference, and ambient pressures.

Results of the mass-transfer tests[7] were prepared for study by plotting water-flow rate vs. protected-surface area, test-section pressure drop vs. protected-surface area, and test-section pressure drop vs. liquid-flow rate. Curves with abruptly changing slopes were drawn through each set of points corresponding to a given set of gas-stream conditions.

[7] Data obtained during the film-stability and evaporation-rate studies described in this paper are presented in tabular form in Tables I through V of Ref. 2.

For the direct study of the character of liquid wall films, a Lucite test section was installed in the test apparatus; study media included high-speed motion pictures, spark photographs, and visual observations aided by stroboscopic lighting.

High-speed motion pictures were taken for gas-stream-diameter Reynolds numbers from 239,000 to 664,000, water-flow rates from 0.031 to 0.12 lb/sec/ft of tube circumference, ambient pressures, and ambient temperatures. The pictures make manifest the following facts for the range of variables investigated: 1 Small disturbances with wave lengths of the order of 10 film thicknesses are present on the surface of the liquid film for all liquid-flow rates. 2 The scale of the small disturbances decreases as the diameter Reynolds number of the gas stream increases; the scale does not vary appreciably, however, when the liquid-flow rate is changed. 3 For liquid-flow rates larger than some critical value, long wave-length disturbances appear on the surface of the film. 4 The inception point of the long wave-length disturbances is independent of the gas-stream Reynolds number. 5 Liquid droplets are entrained by the gas stream from the crests (regions where relatively large quantities of liquid are collected) of the long wave-length disturbances.

Spark photographs were taken for gas-stream-diameter Reynolds numbers from 310,000 to 560,000, water-flow rates from 0.021 to 0.097 lb/sec/ft of tube circumference, ambient pressures, and ambient temperatures. Inspection of the original negatives confirmed items 1 through 4 of the preceding paragraph and presented additional detailed qualitative information concerning the structure of the disturbances. (These spark photographs, incidentally, do not reveal any film nonuniformities which may be attributed to the fact that the film was injected through discrete holes. This state of affairs is good; circumferential film nonuniformities are undesirable in most film-cooling applications.)

Visual observations aided by stroboscopic lighting were made for gas-stream-diameter Reynolds numbers from 307,000 to 595,000, test-section pressures from 14.1 to 28.9 psia, viscosity ratios (μ_{M_0}/μ_{L_f}) from 0.0052 to 0.0140 (obtained by using water and aqueous sucrose solutions), and ambient temperatures. For given gas-stream conditions and given fluid properties, the liquid-flow rate corresponding to the observed inception point of unstable liquid-film flow was determined in nine separate trials. The median of the flow rates corresponding to the nine trials was accepted for comparison with data obtained by other study mediums.

C *Discussion of Data*

Flows with Heated Gases: The abrupt changes in the slopes of the curves of protected-surface area vs. water-flow rate (one abrupt change for each curve) are interpreted to mean that two different types of flow are encountered in film-cooling applications. One type, found at relatively low liquid-flow rates, leads apparently to low mass-transfer rates, whereas the other type, found at relatively high liquid-flow rates, leads apparently to high mass-transfer rates. Obviously, the latter type of flow is to be avoided if efficient coolant usage is desired.

The fact that the curves of test-section pressure drop vs. liquid-flow rate (or vs. protected-surface area) possess abrupt changes in slope which are related to film-attachment efficiency and film stability can be accounted for if one examines the equations of steady one-dimensional gas flow as developed by Shapiro and Hawthorne (12). For further discussion of this topic, see Appendix A of Ref. 2.

Inspection of available data (including those presented by Kinney in Fig. 4 of Ref. 13) for the flow of heated air over a thin water film reveals that data corresponding to the inception point of unstable liquid-film flows (flows which are accompanied by high mass-transfer rates) may be presented in dimensionless form (6) by plotting the dimensionless film

thickness corresponding to the inception point of unstable liquid-film flows as a function of the ratio of the gas-vapor-mixture viscosity to the liquid viscosity; the dimensionless film thickness is defined by $\eta^* = \rho_{Lf}\sqrt{\tau_0/\rho_{Lf}}\,(\eta/\mu_{Lf})$ and (assuming a linear velocity profile to exist in the liquid film) is calculated from $\eta^* = \sqrt{2\Gamma/\mu_{Lf}}$, where Γ is liquid flow per unit time and per unit length of tube circumference, η is average film thickness, the subscript *Lf* refers to the liquid in the film, and the viscosities are evaluated at the liquid-film surface temperature. The film temperatures were calculated using Equation [39], and the mixture viscosities were calculated using the procedure suggested by Bromley and Wilke (14).

Since the liquid viscosity appears in both the abscissa and the ordinate of Fig. 6 and the gas-vapor-mixture viscosity did not vary appreciably for the tests, it is interesting to replot the data after dividing the ordinate by the viscosity ratio (Fig. 7). Further comments on Figs. 6 and 7 are withheld until later in the discussion.

Flows with Unheated Gases: Data for gas flows at ambient temperatures have been added to Figs. 6 and 7; the film temperature was assumed to be equal to the wet-bulb temperature in the case of a water film and to the temperature (measured) of the exhausted liquid in the case of a film of aqueous sucrose solution.

The data presented in Figs. 6 and 7 are for gas-stream-diameter Reynolds numbers from 105,000 to 2,900,000; duct diameters from 2 to 4 in.; test-section pressures from 14.1 to 28.9 psia; temperature ratios T_∞/T_0 from 1.0 to 3.5; and mass-flow ratios $\dot{m}_0/\rho_\infty u_\infty$ from zero to 0.0015 (where T is temperature, and m is mass transfer per unit area and per unit time). Since the viscosity ratio μ_{M_0}/μ_{Lf} (where subscript M refers to the mixture of gases) did not vary monotonically with any of these parameters, the curves presented may be taken to be general curves which are valid for liquid-gas combinations other than those investigated (provided, of course, that the several parameter values do not vary too greatly from the ranges investigated).

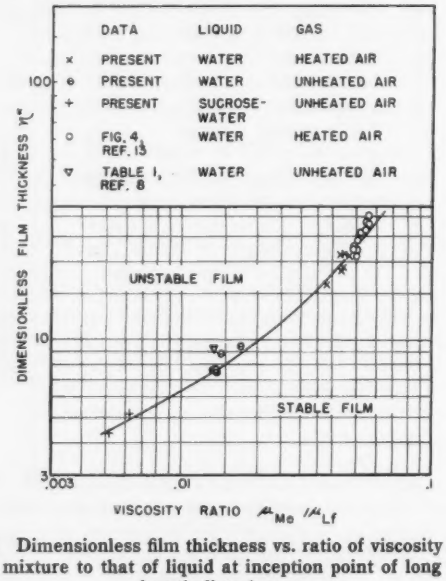


Fig. 6 Dimensionless film thickness vs. ratio of viscosity of gas vapor mixture to that of liquid at inception point of long wave-length disturbances

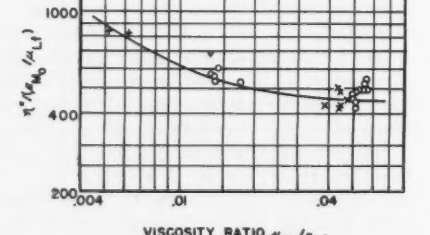


Fig. 7 Dimensionless film thickness η^* divided by μ_{M_0}/μ_{Lf} ($= \rho_{Lf}\sqrt{\tau_0/\rho_{Lf}\,\eta/\mu_{M_0}}$) vs. μ_{M_0}/μ_{Lf} at inception point of long wave-length disturbances

The hypothesis that an annular liquid film flowing under the influence of a high-velocity gas stream behaves as a part of a single-phase boundary layer is now seen to be incorrect. Whereas the laminar sublayer thickness in a single-phase boundary layer is defined completely by the dimensionless thickness y^*, Fig. 6 indicates that the maximum allowable thickness of a stable liquid wall film flowing under the influence of a high-velocity gas stream depends upon both the dimensionless thickness η^* and the viscosity ratio μ_M/μ_{Lf}. The basis for this observed dissimilarity has not been established; it is suspected, however, that the velocity profile as influenced by the viscosity discontinuity at the liquid-gas interface has an appreciable effect on the stability of the liquid wall film (see Fig. 8 for a sketch of a velocity profile typical of that encountered in the case of a stable liquid wall film flowing under the influence of a turbulent gas stream).

Although dimensionless parameters (e.g., the ratio η/δ) other than those used in Figs. 6 and 7 have been examined, only those used in Figs. 6 and 7 were found to be satisfactory for data presentation.

Summarizing, examinations of data concerning mass transfer from liquid wall films, inspections of high-speed motion pictures and spark photographs of liquid wall films, and visual observations of liquid wall films led to the conclusion that the presence of unstable long wave-length disturbances on a liquid-film surface was accompanied by relatively high mass-transfer rates (due, at least in part, to the loss of liquid droplets from the unstable film surface to the gas stream); hence, the unstable long wave-length disturbances are to be avoided when designing for an efficient film-cooling system. All available data corresponding to the inception point of instability for liquid wall films flowing under the influence of high-velocity gas streams are presented in dimensionless form by plotting the dimensionless film thickness corresponding to the inception point of stability as a function of the ratio of the gas-vapor-mixture viscosity to the liquid viscosity, where the viscosities were evaluated at the liquid-film surface temperature.

The dimensionless parameters which were used in the presentation of the data obtained at the inception point of film instability (Figs. 6 and 7) are speculative inasmuch as the liquid density ρ_{Lf} and gas viscosity μ_{M_0} were not varied appreciably during the tests. Experiments with various liquid densities and various gas viscosities are required for the com-

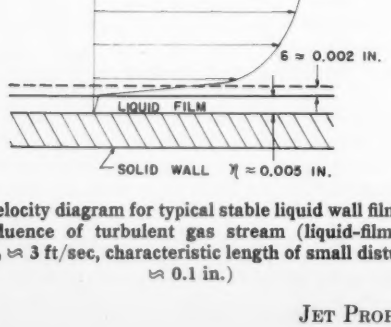


Fig. 8 Velocity diagram for typical stable liquid wall film flowing under influence of turbulent gas stream (liquid-film surface velocity $u_0 \approx 3$ ft/sec, characteristic length of small disturbances ≈ 0.1 in.)

plete confirmation of the dimensionless parameters used in Figs. 6 and 7.

Since the small disturbances which have been observed on the surface of a liquid film flowing under the influence of a turbulent gas stream appear to be related to the gas-stream turbulence, a complete analytical investigation of their origin would be difficult. However, much useful information could perhaps be obtained from an analysis of the stability of Couette flow with two layers of fluid of different densities and viscosities. Such a flow might be stable to small oscillations for all Reynolds numbers, but it is possible that the wave lengths and velocities of the least-damped oscillations are related closely to the small disturbances observed on the surface of a liquid film. The fact that the wave length of the small surface disturbances has been observed to be approximately 10 film thicknesses makes the suggested analysis appear promising; a typical result of stability analyses is that the least-damped oscillations have a wave length of the order of ten times the characteristic length of the flow field.

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(*End of Part 1.*) Part 2 will appear in the next issue of JET PROPULSION.

A New Supply System for Satellite Orbits—Part 2

(*Continued from page 373*)

The author wishes to make the following revisions to Part 1 in the September-October issue of JET PROPULSION:

Page 306: In Equation 10 the first two terms on the right-hand side should be under a common square root sign.

Page 308: In Table 1, all weights designated by W_p should be W_A, since W_α is included in W_n.

Page 309: In first column, line 39 should read: "The optimum distance from Mars is 3525 n. mi., rather than 540 n. mi., i.e., at an altitude of 1695 n. mi." In second column, the last sentence should read: "Payloads of 200, 1200, and 11,000 lb, respectively, for these three vehicles were found to be a reasonable tentative base for the establishment of a small, inhabited satellite of 20,000 cu ft volume, . . .'"

Transpiration Cooling in Porous Metal Walls

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The internal heat-transfer equations (1)[3] for a porous medium under transpiration cooling are simplified by approximations valid for porous metals. The temperature distribution problem for porous metal walls is reformulated on the basis of over-all energy balance requirements (2). The simplified formulation is applied to plane parallel and cylindrical walls subject to boundary conditions corresponding to convective heat transfer and specified temperatures at the wall boundaries.

Nomenclature

c_p = specific heat at constant pressure
d = pore diameter
h = coefficient of heat transfer
l = wall thickness
m = $r_g G_{c_g} c_p / K$
n = number of pores per unit area
r = radius
x = position co-ordinate in wall
C = constant of integration
G = mass flux
K = thermal conductivity of porous material
N = h/Gc_p
T = temperature
η = ratio of temperature differences
λ = Gc_p/K

Subscripts

c = coolant or coolant side
g = gas or gas side
i = internal
w = wall

1 Introduction

THE differential equations governing internal heat transfer and temperature distributions within a porous medium under transpiration cooling have been formulated in (1) and (3). In the former reference, Weinbaum and Wheeler have obtained analytic results relating the wall matrix and coolant temperature distributions, $T_w(x)$ and $T_c(x)$, to specified temperature values at the wall boundaries. In engineering applications, the use of such analytic results has been limited because of their mathematical complexity. For porous metal walls the mathematical complications may be avoided by a reformulation of the internal heat flow equations on the basis of the approximate equality of the matrix and coolant temperatures within the wall, $T_w(x) \approx T_c(x)$, which is a consequence of the large area available for heat transfer between the coolant and the matrix (2). This approximate equality fails to hold only in the immediate neighborhood of the coolant side boundary extending about one pore diameter into the interior (1).

The simplified formulation of the transpiration cooling problem based on $T_w(x) \approx T_c(x)$ is applied in this paper to the derivation of temperature distributions associated with given boundary conditions corresponding to convective heat transfer and fixed temperatures at the wall boundaries. Formulas are obtained for both plane parallel and cylindrical wall configurations.

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[3] Numbers in parentheses refer to References on page 378.

2 Simplified Analysis

A The Differential Equations of Internal Heat Transfer

In their analysis, Weinbaum and Wheeler (1) consider a model of the porous wall formed by identical cylindrical pores of diameter d running parallel through a wall of thickness l. Internal heat flow along the wall matrix occurs with an effective thermal conductivity K, and heat transfer from the matrix to the coolant is described by a convection law with uniform internal convection coefficient h_i acting on the temperature difference $T_w(x) - T_c(x)$. Thus

$$K \frac{d^2T_w}{dx^2} = \pi n d h_i [T_w - T_c] \quad \ldots\ldots [1]$$

where n is the number of cylindrical pores per unit area of the wall. The coolant flowing in the negative x-direction (see Fig. 1) absorbs the heat transferred from the matrix, and its temperature, therefore, satisfies the relation

$$-G_c c_p \frac{dT_c}{dx} = K \frac{d^2T_w}{dx^2} \quad \ldots\ldots [2]$$

where G_c is the coolant mass flux and c_p is the coolant specific heat at constant pressure.

Analogous relations for transpiration cooling in walls of cylindrical configuration with radial coolant flow are

$$K \frac{d}{dr} r \frac{dT_w}{dr} = \pi n d h_i [T_w(r) - T_c(r)] \cdot r \quad \ldots\ldots [1']$$

$$-rG_c(r)c_p \frac{dT_c}{dr} = K \frac{d}{dr} r \frac{dT_w}{dr} \quad \ldots\ldots [2']$$

In the approximation that the coolant and matrix temperatures are approximately equal within the porous metal wall, we may set

$$T = T_w \approx T_c \quad \ldots\ldots [3]$$

and obtain for the plane wall

$$-G_c c_p \frac{dT}{dx} = K \frac{d^2T}{dx^2} \quad \ldots\ldots [4]$$

while for the cylindrical wall

$$-rG_c c_p \frac{dT}{dr} = K \frac{d}{dr} r \frac{dT}{dr} \quad \ldots\ldots [4']$$

B Boundary Conditions

In (1) the solution of the system of differential Equations [1], [2] or [1′], [2′] is obtained by the elimination of either T_w or T_c leading to a single third-order differential equation for the remaining dependent variable which is then completely determined by three boundary conditions, such as the specification of matrix boundary temperatures T_{wg}, T_{wc}, and

the coolant entrance temperature T_{cc}. For the simplified formulation given by Equation [4] or [4′], the second-order equation requires the statement of only two boundary conditions. Thus, with constant temperature conditions at the boundaries, T_{wg} and T_{wc}, the matrix temperature, $T_w = T_c$, is completely determined from [4] or [4′]. Furthermore, T_c is also given by the same form except for the immediate neighborhood of the coolant boundary where T_c changes from the value T to the value T_{cc} in a short distance of about one pore diameter. This rapid change in T_c, which is of little engineering interest, complicates the analytic expressions obtainable from Equations [1], [2] or [1′], [2′].

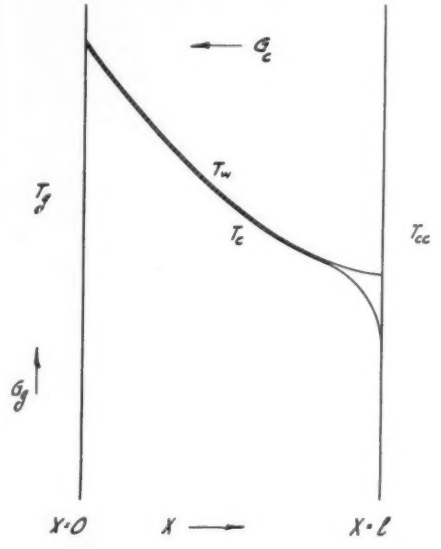


Fig. 1 Temperature distribution in cooled porous wall

A class of boundary conditions of considerable interest in engineering applications consists in the description of convective heat transfer at the porous wall boundaries. With reference to the plane wall sketched in Fig. 1, such boundary conditions may be specified by

$$h_g[T_g - T_w] = -K\frac{dT_w}{dx} \qquad \text{at } x = 0 \quad \ldots\ldots [5]$$

$$h_c[T_w - T_{cc}] = -K\frac{dT_w}{dx} \qquad \text{at } x = l \quad \ldots\ldots [6]$$

where h_g is the effective heat transfer coefficient at the "gas-side" boundary, and h_c is the coefficient at the coolant boundary. The third boundary condition needed to determine T_w and T_c from Equations [1], [2] is

$$T_c = T_{cc} \qquad \text{at } x = l \quad \ldots\ldots [7]$$

In the simplified formulation of Equations [3] and [4], only two boundary conditions are needed. The first of these is a statement of the heat flux continuity at the gas-side boundary

$$h_g[T_g - T] = -K\frac{dT}{dx} \qquad \text{at } x = 0 \quad \ldots\ldots [8]$$

The second condition is the over-all heat balance equation

$$h_g[T_g - T]_{x=0} - h_c[T - T_{cc}]_{x=l} = G_c c_p[T - T_{cc}]_{x=0} \quad \ldots\ldots [9]$$

stating that the excess of the heat flux entering on the "gas-side" over that leaving the wall at the coolant boundary appears as the rate of increase in the coolant enthalpy in transit through the wall.

For the cylindrical wall bounded by the radii r_g, r_c, analogous convective boundary conditions to be used with Equation [4′] are

$$r_g h_g[T_g - T] = -r_g K\frac{dT}{dr} \qquad \text{at } r = r_g \quad \ldots\ldots [8']$$

$$r_g h_g[T_g - T(r_g)] - r_c h_c[T(r_c) - T_{cc}] = r_g G_{cg} c_p[T(r_g) - T_{cc}] \quad \ldots\ldots [9']$$

where the coolant mass flux G_{cg} is referred to the inner radius $r = r_g$. It is to be noted that the mass flux G_c is dependent here on the radius r within the wall, in accordance with the mass flow continuity law

$$rG_c(r) = r_g G_{cg} \quad \ldots\ldots [10]$$

3 Applications

A The Plane Parallel Wall

The general solution for the temperature within the plane parallel wall is obtained from Equation [4] as

$$T(x) = C_1 + C_2 e^{-\lambda x} \quad \ldots\ldots [11]$$

where C_1 and C_2 are integration constants and

$$\lambda = \frac{Gc_p}{K} \quad \ldots\ldots [12]$$

a Constant-Temperature Boundary Conditions

When the integration constants C_1 and C_2 are determined from the constant-temperature boundary conditions

$$T_{wg} = T(0) = C_1 + C_2$$
$$T_{wc} = T(l) = C_1 + C_2 e^{-\lambda l}$$

the expression for the internal temperature distribution becomes

$$T(x) = \frac{T_{wg}(e^{-\lambda x} - e^{-\lambda l}) + T_{wc}(1 - e^{-\lambda x})}{1 - e^{-\lambda l}} \quad \ldots\ldots [13]$$

For the porous metal data employed in reference (1), the simplified result in Equation [13] yields sensibly the same matrix temperatures as the more complicated expression of that reference. Similarly, the coolant temperature is also accurately described by Equation [13] except, as remarked above, for the immediate neighborhood of the coolant boundary where the coolant temperature changes from T_{cc} to nearly T_{wc} in about one pore diameter.

b Convective Boundary Conditions

With the convective boundary conditions in Equations [8] and [9], the integration constants in the general solution, Equation [11], are determined from

$$h_g[T_g - (C_1 + C_2)] = K\lambda C_2 = G_c c_p C_2$$
$$h_g[T_g - (C_1 + C_2)] - h_c[(C_1 + C_2 e^{-\lambda l}) - T_{cc}] = G_c c_p[(C_1 + C_2) - T_{cc}]$$

with the result

$$C_1 = -\left[\frac{N_g N_c e^{-\lambda l}}{(N_g + 1)(N_c + 1)} T_g - T_{cc}\right] \times \left[1 - \frac{N_g N_c e^{-\lambda l}}{(N_g + 1)(N_c + 1)}\right]^{-1} \quad \ldots\ldots [14]$$

$$C_2 = \frac{N_g}{N_g + 1}[T_g - T_{cc}]\left[1 - \frac{N_g N_c e^{-\lambda l}}{(N_g + 1)(N_c + 1)}\right]^{-1} \quad \ldots\ldots [15]$$

where $N_g = h_g/G_c c_p$ and $N_c = h_c/G_c c_p$.

A basic engineering parameter of interest in transpiration cooling is the so-called efficiency η_1, defined by the ratio

$$\eta_1 = \frac{T_g - T_{wg}}{T_g - T_{cc}} = \frac{T_g - (C_1 + C_2)}{T_g - T_{cc}}$$

which, for the convective boundary conditions, becomes

$$\eta_1 = \frac{1 + G_c c_p/h_c}{1 + \frac{G_c c_p}{h_c} + \frac{h_g}{h_c} + \frac{h_g}{G_c c_p}\left(1 - e^{-\frac{G_c c_p l}{K}}\right)} \quad \ldots\ldots [16]$$

Another parameter of interest involves the temperature difference across the porous wall

$$\eta_2 = \frac{T_{wg} - T_{wc}}{T_g - T_{cc}} = \frac{C_2(1 - e^{-\lambda l})}{T_g - T_{cc}}$$

which, for the convective boundary conditions, may be reduced to

$$\eta_2 = \frac{\frac{h_g}{G_c c_p}\left(1 + \frac{G_c c_p}{h_c}\right)\left(1 - e^{-\frac{G_c c_p l}{K}}\right)}{1 + \frac{G_c c_p}{h_c} + \frac{h_g}{h_c} + \frac{h_g}{G_c c_p}\left(1 - e^{-\frac{G_c c_p l}{K}}\right)} \quad \ldots.[17]$$

For vanishingly small coolant flow

$$G_c \to 0, \qquad e^{-\frac{G_c c_p l}{K}} \to 1 - \frac{G_c c_p l}{K}$$

the above results for η_1 and η_2 reduce to well-known over-all temperature relations

$$\eta_1 = \frac{T_g - T_{wg}}{T_g - T_{cc}} \to \frac{1}{1 + h_g/h_c + h_g l/K}$$

$$\eta_2 = \frac{T_{wg} - T_{wc}}{T_g - T_{cc}} \to \frac{h_g l/K}{1 + h_g/h_c + h_g l/K}$$

obtainable directly from heat balance equations for impermeable walls

$$h_g[T_g - T_{wg}] = K\frac{T_{wg} - T_{wc}}{l} = h_c[T_{wg} - T_{cc}]$$

Another limiting relation of practical importance is obtainable from Equation [16] for $h_c \to 0$, inasmuch as convection and radiation losses through the coolant side boundary may be neglected under typical transpiration cooling test conditions

$$\eta_1 \to \frac{1}{1 + h_g/G_c c_p} \qquad \text{for } h_c \to 0$$

It is to be noted that η_1 here becomes independent of the matrix data K and l and that, in so far as the gas-side heat-transfer coefficient is proportional to the gas mass flux G_g, empirical correlations (4) between η_1 and ratio G_c/G_g can be justified analytically.

An empirical approach to the study of the relation between the effective gas-side coefficient h_g in the presence of injection and the coefficient $h_g°$ obtainable without coolant injection is suggested by an analytic formula for the ratio η_2/η_1 taken directly from Equations [16] and [17]

$$\frac{\eta_2}{\eta_1} = \frac{T_{wg} - T_{wc}}{T_g - T_{wg}} = \frac{h_g}{G_c c_p}\left(1 - e^{-\frac{G_c c_p l}{K}}\right) \quad \ldots\ldots [18]$$

This ratio is independent of heat losses through the coolant side boundary as well as of entrance coolant temperature and is, therefore, free from some possible sources of error (4, 5) in the interpretation of test data.

B Cylindrical Walls

Referring the coolant flux $G_c(r)$ to the inner or gas-side boundary of the circular cylinder by means of Equation [10], we have the general solution of Equation [4′]

$$T(r) = C_1' + \frac{C_2'}{r^m} \quad \ldots\ldots [11']$$

where C_1' and C_2' are integration constants and

$$m = \frac{r_g G_{cg} c_p}{K} \quad \ldots\ldots [12']$$

a Constant-Temperature Boundary Conditions

When C_1 and C_2 are determined from the constant-temperature boundary conditions

$$T_{wg} = T(r_g) = C_1' + \frac{C_2'}{r_g^m}$$

$$T_{cc} = T(r_c) = C_1' + \frac{C_2'}{r_c^m}$$

the expression for the internal temperature distribution becomes

$$T(r) = \frac{T_{wg}\left[\left(\frac{r_g}{r}\right)^m - \left(\frac{r_g}{r_c}\right)^m\right] + T_{wc}\left[1 - \left(\frac{r_g}{r}\right)^m\right]}{1 - \left(\frac{r_g}{r_c}\right)^m} \quad \ldots [13']$$

The result obtained here agrees with the more general expression for the temperature distribution in the cylinder derived in reference (1) when the latter expression is simplified on the basis of numerical data characterizing transpiration cooling in porous metal walls.

b Convective Boundary Conditions

With the convective boundary conditions in Equations [8′] and [9′] the integration constants in Equation [11′] are determined from

$$h_g\left[T_g - \left(C_1' + \frac{C_2'}{r_g^m}\right)\right] = Km\frac{C_2'}{r_g^{m+1}} = G_{cg}c_p\frac{C_2'}{r_g^m}$$

$$r_g h_g\left[T_g - \left(C_1' + \frac{C_2'}{r_g^m}\right)\right] - r_c h_c\left[\left(C_1' + \frac{C_2'}{r_c^m}\right) - T_{cc}\right] = r_g G_{cg} c_p\left[\left(C_1' + \frac{C_2'}{r_g^m}\right) - T_{cc}\right]$$

with the result

$$C_1' = -\left[\frac{N_g' N_c'(r_g/r_c)^m}{(N_g' + 1)(N_c' + 1)}T_g - T_{cc}\right] \times \left[1 - \frac{N_g' N_c'(r_g/r_c)^m}{(N_g' + 1)(N_c' + 1)}\right]^{-1} \quad \ldots. [14']$$

$$C_2' = \frac{N_g' r_g^m}{N_g' + 1}[T_g - T_{cc}]\left[1 - \frac{N_g' N_c'(r_g/r_c)^m}{(N_g' + 1)(N_c' + 1)}\right]^{-1} \quad \ldots\ldots [15']$$

where $N_g' = h_g/G_{cg}c_p$ and $N_c' = h_c/G_{cg}c_p$.

With these results the "efficiency" parameters for convective boundary conditions become

$$\eta_1 = \frac{T_g - T_{wg}}{T_g - T_{cc}} = \frac{1 + \frac{G_{cg}c_p}{h_c}}{1 + \frac{G_{cg}c_p}{h_c} + \frac{h_g}{h_c} + \frac{h_g}{G_{cg}c_p}\left[1 - \left(\frac{r_g}{r_c}\right)^m\right]} \quad \ldots\ldots [16']$$

$$\eta_2 = \frac{T_{wg} - T_{wc}}{T_g - T_{cc}} = \frac{\frac{h_g}{G_{cg}c_p}\left(1 + \frac{G_{cg}c_p}{h_c}\right)\left[1 - \left(\frac{r_g}{r_c}\right)^m\right]}{1 + \frac{G_{cg}c_p}{h_c} + \frac{h_g}{h_c} + \frac{h_g}{G_{cg}c_p}\left[1 - \left(\frac{r_g}{r_c}\right)^m\right]} \quad \ldots\ldots [17']$$

$$\frac{\eta_2'}{\eta_1'} = \frac{T_{wg} - T_{wc}}{T_g - T_{wg}} = \frac{h_g}{G_{cg}c_p}\left[1 - \left(\frac{r_g}{r_c}\right)^m\right] \quad \ldots\ldots [18']$$

The latter expression permits the empirical determination of

(*Continued on page 378*)

A New Supply System for Satellite Orbits—Part 2[5]

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The subject of orbital supply is investigated, stressing the engineering possibilities for improving the economic aspects of orbital operations and space flight. Following a brief survey of the basic problems, a new orbital supply system is proposed in the second section of this paper. It is characterized by operational separation of cargo supply and passenger transport. The supply ship has a ballistic shape and is automatically guided into the target orbit. The passenger ship is equipped with a winged upper stage. In the third section, supply requirements for orbital operations as well as space flight are discussed. The existence of optimum satellite orbits for departure and arrival of interplanetary expeditions is demonstrated analytically. Finally, examples of guided supply vehicles and passenger ships are presented. Their take-off weight and payload capacity serve to illustrate the effect of supply requirements on the over-all supply efforts.

Orbital Vehicles

It is the purpose of this section to present possible applications of the general principles outlined in the second section, and to illustrate the economic advantage of the proposed supply system.

Numerical data, given in this connection, should be regarded as examples to point out the order of magnitude or differences in trends. Considerable efforts have been made, however, to avoid conjecture wherever possible, although evidence cannot always be submitted in this paper, due to lack of space. Nevertheless, the data and design assumptions are, of course, open to argument, whereas the trends are considered much more certain. A more detailed discussion of the vehicles, in conjunction with the small satellite, is planned in a subsequent paper.

The 11,000-lb Supply Ship

In point 2 at the end of Section 2, attention was called to the fact that operational separation of the supply service would permit each of the two basic configurations to be designed for its particular purpose. The necessity to compromise is greatly reduced in comparison to winged supply vehicles. Better economy in the use of expensive materials and greater design efficiency are the result.

In Fig. 8, a supply ship configuration is depicted which appears promising for a minimum dead weight design and delivery of a maximum of useful material with each flight.

The concept involves a three-stage vehicle where the third stage is partly surrounded by the second stage. The containers of stage 3 are protected while passing through the atmosphere and will not emerge until practically vacuum conditions are reached. They can be made of very thin material and are arranged between round, tubular columns which serve as thrust frame, connecting motor section and payload nose. Assembly of the two upper stages appears well feasible. Accessibility to all important parts is good, since the propulsion system is in line with that of the second stage, while the electronic equipment, that is, guidance, control, power source and attitude control, are in the nose section. The power plant operates jointly with that of the second stage, fed by the latter, until they separate. Thereafter the third stage continues alone. Thereby power plant weight is saved in the second stage. The telescope arrangement of the two upper stages reduces the over-all length and thus facilitates handling and launching.

A tentative weight analysis of this vehicle is based on the power plant and flight performance data assembled in Table 3. Table 4 summarizes the essential component weight ratios and some main weight data for the 11,000-lb supply ship. These ratios, as well as those presented in subsequent tables are the result of studies using published weight data and unclassified theoretical analyses, such as reference (10). Values were rounded off in several cases. The symbols are explained in the nomenclature at the beginning of this paper.

As Table 4 indicates, the joint thrust force of both upper stages is so distributed that stage 3 always has a certain weight with respect to the second stage (cf. the corresponding initial axial load factors in Table 3). This weight increases during the burning time, as stage 2 is emptied, while stage 3 remains fully loaded. The payload nose of stage 3 rests on a conical bearing on top of stage 2 so that lateral displacements due to inertial forces are also avoided. Because of the weight of stage 3, premature separation should not occur. At burn-out of the second stage, the thrust of the third stage produces separation.

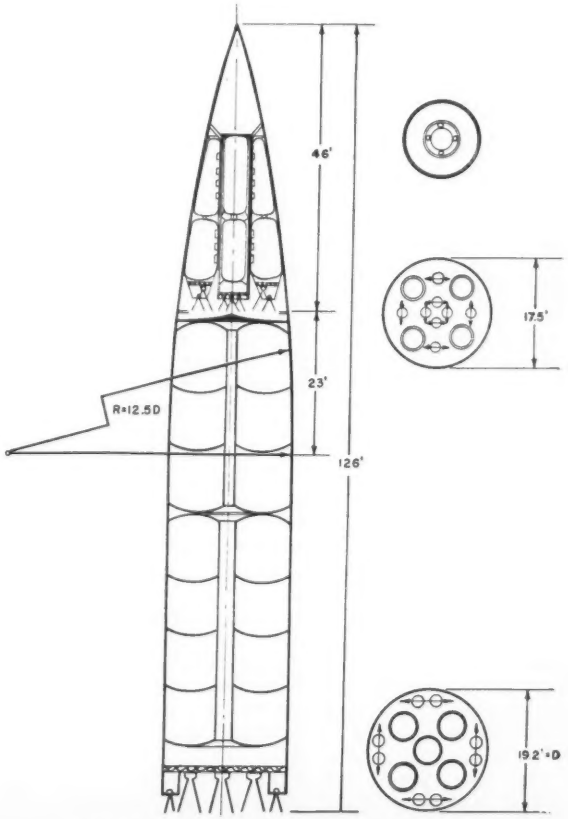


Fig. 8 Schematic sketch of 11,000-lb supply ship

[1] Preliminary Design Department. Mem. ARS.

[5] Part 1 appeared in the September-October issue of JET PROPULSION, pp. 302–309.

Table 3 Trajectory Data for Powered Ascent of Supply Vehicles (Burning Period I)

Stage number	1	2	3
Propellant		Oxygen-Hydrazine	
Mixture ratio (oxid/fuel, b.w.)		0.9	
Chamber pressure, psia		590	
Exit pressure, psia	10.7	1.47	1.47
Theoret. spec. impulse, $I_{sp\,th}$, sec	305	350	350
Velocity correction factor, $I_{sp}/I_{sp\,th}$	0.96	0.93	0.93
Used spec. impulse, I_{sp}, sec	295	325	325
Effective loading factor, W_A/W_0	0.695	0.642	0.581[a]
Effective mass ratio (−)	3.28	2.8	2.39[b]
Cut-off trajectory angle, deg	15	4	0
Initial axial load factor, g	1.3	1.5	1.44
Axial load factor at cut-off g	4.26	4.19	3.46
Maximum normal load factor (deflection), g	0.33	0.79	0.62
Velocity at cut-off, fps	8,060	18,170	25,600
Altitude at cut-off, 1000 ft	147	366	435
Burning time, sec	155	139.1	104
Actual velocity increment, Δv, fps	8,060	10,110	7,430
Ideal velocity increment, Δv_{id}, fps	11,000	10,700	7,500
Stage cut-off velocity factor, $\Delta v/\Delta v_{id}$	0.735	0.947	0.99
Over-all ideal velocity, v_{id}, fps (all burning periods)		31,000	
Over-all cut-off velocity factor v_l/v_{id}		0.89	
Over-all mass ratio (−)		22.3	

[a] Total: 0.591
[b] Total: 2.44

The Passenger Ship

In designing the upper stage of the passenger ship, the most important viewpoints are good hypersonic flight qualities, reliability, and safety. These considerations suggest a shape similar to that shown in Fig. 9, for example. The vehicle consists of three basic components, nose section (a), wings (b), and main body (c). The nose section contains the passenger space with all furnishings, power source and guidance and stability equipment. This section consists of a cylindrical body with a nose cone obliquely attached to it, so that the bottom meridian line is rectilinear. It is embedded in the wing portion, preceding the main body. The latter contains essentially three complete rocket power plants, mounted parallel to each other. Thereby the vehicle body (a) and (c) obtains two-dimensional shape with a large flat bottom area which provides the bulk of the required lift in the hypersonic and high supersonic flow regime, directly supporting a large portion of its own weight. The wings (b) furnish additional lift and, because their aspect ratio is much larger than that of the body, reduce the angle of attack for low-speed flight and landing. Front and rear landing skids are provided. The front skid is a plate which can serve as auxiliary wing, tiltable to assist the ailerons in maintaining the desired angle of attack at low speed. Wedge-shaped trimmers at the stern (so as not to interrupt the flat bottom surface) provide pitch control at hypersonic speed to maintain adequate angles of attack for descent at maximum lift coefficient, thereby reducing frictional heating. For certain emergency conditions, separation of the passenger section (a) from the main body is provided, as indicated by the dashed line in Fig. 8. By this method, ejection in individual capsules (11), with its inherent complexity, hence increased probability of failure, and high weight can be avoided. The mechanism and aerodynamics of separation cannot be discussed in this paper. Once separated, this section is essentially a delta-wing plane, equipped with its own aerodynamic control. It can glide to the ground, or at least carry the passengers back to lower altitude where they can bail out.

The passenger plane must have booster stages. If flat-bottom bodies are involved, a parallel arrangement of the two upper stages appears promising.

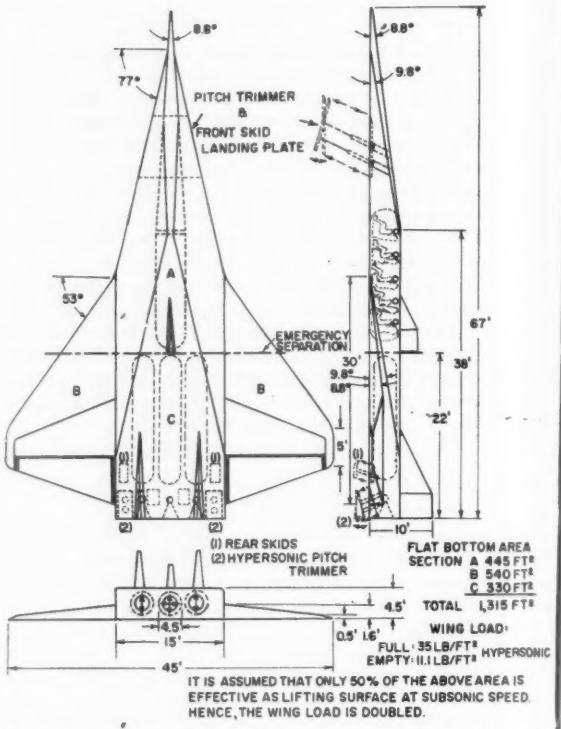


Fig. 9 Schematic sketch of 5-person orbital rocket glider

Design concepts of this type have been studied since 1951[6] and ultimately lead to configurations of which an example is depicted in Fig. 10[7], to serve as a prototype for the present discussion. The advantages of the twin-stage arrangement are: reduced over-all height (simpler handling), improved stability, and saving of power plant weight due to joint burning of the twin stages prior to separation.[8]

Weight studies indicated that the over-all take-off weight of the passenger ship can be made approximately equal to that of the 11,000 lb supply ship, if the second stage is made expendable, that is, not provided with wings. The corresponding weight data are presented in Table 4. The gross weight of the passenger stage equals that of the 11,000-lb payload stage, at a payload, however, of 1200 lb only. The second stage is larger than in the supply ship. Its structural weight is somewhat smaller, since part of its structure is protected by the passenger ship, and, therefore can be light. The first stage is smaller and has larger structural weight due to the fins required for stabilization. The over-all mass ratio of supply ship and passenger vehicle is 28.5 and 32.4, respectively.

[6] Thanks are due to Mr. Hamilton, Propulsion and Fuels Branch, Army Guided Missile Laboratories, Redstone Arsenal, Huntsville, Ala., for his valuable help during the first phase of these design studies.

[7] There are other promising configurations conceivable. By separating supply and passenger service, the freedom of design is greatly increased.

[8] After completion of this paper it has come to the attention of the author that G. A. Crocco (12) also has considered parallel mounting of stages. Details are not yet known to this author.

Table 4 Component Weight Ratios and Main Weight Data

Vehicle	11,000-lb Supply ship			5-Passenger orbital vehicle		
Stage number	1	2	3	1	2	3
Weight Ratios:						
Payload, λ	0.173	0.198	0.239	0.204	0.168	0.026
Propellant, Λ	0.695	0.642	0.591	0.654	0.682	0.584
Auxiliary fluid, α	0.03	0.025	0.025	0.03	0.030	0.04
Structure, σ	0.065	0.08	0.065	0.075	0.065	0.25
Power plant, δ	0.035	0.05	0.05	0.035	0.05	0.04
GCPE,[a] γ_ϵ	0.002	0.005	0.03	0.002	0.005	0.06
Dead weight, D	0.102	0.135	0.145	0.112	0.120	0.35
Empty weight, E	0.275	0.333	0.389	0.316	0.288	0.376
Design mass ratio, $1/1-(\Lambda+\alpha)$	3.64	3.0	2.61	3.51	3.47	2.66
Net weight factor, f_n	0.160	0.199	0.223	0.173	0.180	0.34
Weights						
Payload weight, $W\lambda$, lb	232,000	46,000	11,000	274,000	46,000	1,200
Effective propellant wt., $W\Lambda$, lb	931,000	148,800	27,300	875,000	186,130	26,900
Dead weight, W_d, lb	137,000	31,300	6,650	157,000	33,000	16,100
Gross weight, W_0, lb	1,340,000	232,000	46,000	1,340,000	274,000	46,000
Miscellaneous:						
Momentum thrust per stage, lb	1,741,000	346,000	66,400	1,741,000	410,000	49,800
No. of motors and thrust,[b] lb	5 at 285,000	4 at 53,500	4 at 16,600	5 at 285,000	4 at 86,000	3 at 16,600
Steering motors,[c] lb	8 at 40,000[d]	4 at 16,600	[e]	8 at 40,000[d]	4 at 16,600[c]	[f]

[a] GCPE = Guidance, Control, Power Equipment; [b] for 2nd stage, thrust of 3rd stage must be added; [c] tiltable, except cf.[f]; [d] 2nd stage motors with low-altitude nozzles; [e] all main motors tiltable for three-axes control; chamber pressure reduced for low-thrust operation at the end of Phase II and during Phase III. [f] central motor swivel-mounted for control in pitch and yaw; side motors tiltable for roll control; vernier thrust produced at the end of Phase II, and during Phases III and IV.

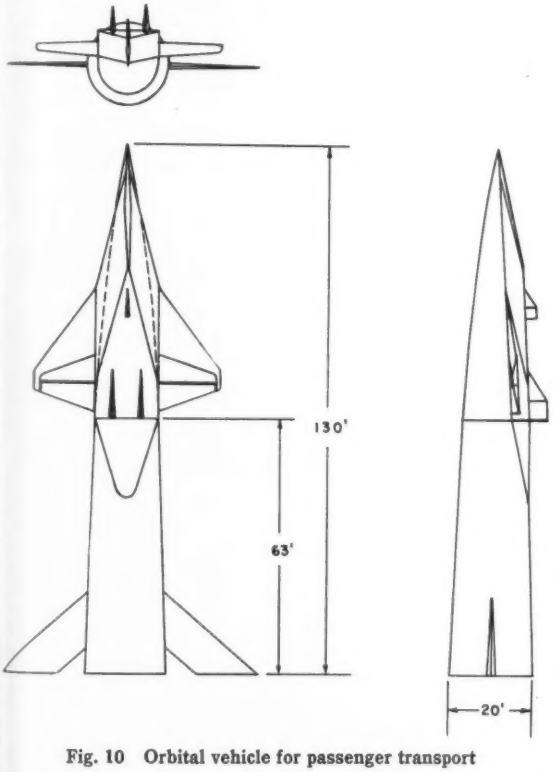


Fig. 10 Orbital vehicle for passenger transport

The 2000-lb Supply Ship

The small supply ship with 2000-lb payload does not offer additional problems. The simplest and most straightforward design is probably a three-stage configuration with tandem arrangement of the two upper stages[9]. The resulting take-off weight is about 275,000 lb (Table 5), placing this ship into a class of multistage vehicles which could be developed within a decade.

Table 5 Component Weights of the 2000-lb Supply Ship

Stage	1	2	3
Payload weight, lb	46,000	8,730	2000
Effective propellant wt., lb	191,000	29,570	5160
Auxiliary fluid wt, lb	8,250	1,150	220
Dead weight, lb	29,750	6,550	1350
Gross weight	275,000	46,000	8730
Design mass ratio $1/1-(\Lambda+\alpha)$	3.64	3.0	2.61
Net weight factor, f_n	0.166	0.207	0.233

Comparison of Vehicles; Supply Economy

For a comparison of the prototype vehicles described before, three parameters are introduced, all of which are based on the delivered payload weight

$$\text{take-off weight factor } \varphi_0 = \frac{W_{01}}{W_{\lambda_3}}$$

where W_{01} is the take-off weight of the first stage, hence, the over-all take-off weight, and W_{λ_3} is the delivered payload weight of the third stage

$$\text{propellant weight factor } \varphi_p = \frac{\Sigma_1^3 W_p}{W_{\lambda_3}}$$

where the propellant comprises the effective propellant used for acceleration and the auxiliary fluid, assuming the use of gas turbines as prime movers, and

$$\text{dead weight factor } \varphi_d = \frac{\Sigma W_d}{W_{\lambda_3}}$$

where the term in the numerator designates the dead weight of all wingless stages. This definition implies that all wingless stages are expendable. Actually there is the alternative of attempting parachute recovery. In reference (13) it has been assumed, on the basis of preliminary calculations, that both first and second stage of a satellite ship can be recovered in this

[9] The most economical solution, employing winged first or second stage (take-off weight about 475,000 lb) is discussed in this author's paper, "Analysis of Orbital Systems," presented at 5th IAF Congress, Innsbruck, Sept. 1954.

manner. However, in view of the intense heat transfer which must be expected on a body of blunt shape and small lift and of the low heat capacity of the thin, unprotected structure, recovery of the second stage appears doubtful. Moreover, remembering that the dead weight of the second stage represents about 14 to 18 per cent of the over-all dead weight and that the flight range of this stage is very long, because of the high cut-off velocity, recovery hardly seems worth while even if the stage could be reconditioned for further use. Recovery of the first stage should be feasible and is more promising, since in that case 75 to 80 per cent of the over-all hardware is gained back. The case of first-stage parachute recovery is considered in the modified dead weight factor φ_d' which then, for the passenger ship, implies that the first and the third stage are recovered after each mission. Table 6 summarizes these parameters for the three prototype vehicles.

Table 6 Supply Economy and Transport Efficiency of Prototype Vehicles

Vehicle	Passenger ship	2000-lb supply ship	11,000-lb supply ship
Type	Winged upper stage	Ballistic	Ballistic
Take-off weight factor			
φ_0	1,120	137.5	122
$1/\varphi_0$	0.000894	0.00729	0.6082
Propellant weight factor			
φ_p	945	117.6	105
$1/\varphi_p$	0.00106	0.0085	0.0095
Dead weight factor			
φ_d	158	18.8	16.0
$1/\varphi_d$	0.0063	0.0532	0.0625
Modified dead weight factor			
φ_d'	27.5	4	3.45
$1/\varphi_d'$	0.0364	0.25	0.29

The take-off weight factor is a measure of the supply economy in general. It can be seen that for the large supply ship this value is better than that of the passenger ship by a factor of about 9. This fact alone, aside from the other advantages pointed out before, characterizes the decisive reduction in over-all efforts which may be gained by the use of automatically controlled supply ships of ballistic shape rather than winged and manned satellite ships. The small supply ship, although greatly superior to the passenger ship, does not quite reach the values obtained with the large vehicle, due to its smaller size.

Propellant and dead-weight factor are measures of the transport efficiency in particular. It is interesting to note on the example of φ_d that, in spite of recovery of its third stage, the passenger ship has a much higher dead-weight factor than the supply vehicles although all their hardware supposedly is lost. In the case of first-stage parachute recovery an excellent transport efficiency is obtained with little difference between the large and the small supply ship, while the passenger ship still loses eight times as much hardware per pound of payload as the 11,000-lb supply ship.

The supply economy, using thermochemical power plants, is very poor in general. Any hope for improving this situation rests with the capability of controlling and utilizing, in time, more powerful energy sources. However, the results compiled in Table 6 serve to show that considerable improvements are more readily obtainable through careful analysis of engineering possibilities which are still far from being exhausted. Assuming parachute recovery of the first stage, a high transport efficiency is obtained even with take-off weight factors of 120 to 140. If, in addition, the orbital and astronautical objectives are selected with proper consideration of the inherent limitations imposed by chemical propellants, then there is no need to wait for more powerful energy sources, nor is there any justification for the assumption that satellite supply or space flight with chemical propellants is preposterously expensive in view of its limited material return.

Over-all Requirements

It is now possible to combine the results obtained in this and the preceding section and to survey the total amount of material required for the establishment of a small satellite, circumnavigation of the moon, and flight to Mars. Based on the weight data for the 11,000-lb supply ship and approximate number of supply flights, Table 7 presents quantity of material consumed if nothing were recovered except the passenger glider. A comparison with the efficiency parameters of the passenger ship indicates that hardware expenses and propellant consumption, respectively, would be at least six times and nine times as large, if ships with winged upper stage were used for cargo transport. This is based on first-stage parachute recovery. The propellants represent the bulk of the weight, as was to be expected, but they are less expensive than the other material. No cost estimates are attempted, because of the future character of such projects. Material requirements, based on generally valid performance calculations and design considerations, reflect the over-all effort more reliably.

Structural and other material actually will not be lost completely. At least, there is use for the third stage in space and parts can be carried back to earth. If one assumes recovery of the first stage, then the propellants will amount to more than 90 per cent of the over-all transport expenses for the respective operation. The same is true for the regular maintenance service. Presumably one or two ascents each of the 2000-lb supply ship and of the passenger ship per month will suffice for this purpose. With parachute recovery the first stage of the passenger ship can also be salvaged, reducing φ_0 to a value of about 27. The propellant consumption would be about 2.8 million pounds for all four flights. The underlying, arbitrary assumption for the two ascents of the passenger ship is that personnel must be rotated every two weeks. This is not necessarily correct, and longer periods of rotation would reduce the number of passenger flights as well as of ascents of

Table 7 Material Requirements for Supply of Operation in Space

(11,000-lb supply ship)

Operation	Establishment of 20,000 cu ft satellite[a]	Circumnavigation of the moon (3 persons)	Mars expedition (8 persons) Case 1	Case 2
No. of supply flights	50	55	680	600
Propellants (O_2—N_2H_4)				
W_A (1000 lb)	55,300	60,808	753,000	663,360
W_α (1000 lb)	2,300	2,550	32,000	27,600
Structure and power plant installation				
W_σ (1000 lb)	5,500	6,050	76,000	67,000
W_δ (1000 lb)	3,050	3,350	41,500	36,600
GCPE[b]				
$W_{\gamma\epsilon}$ (1000 lb)	265	291	3,600	3,180

[a] Four persons; h_s = 540 n.mi.
[b] GCPE = Guidance, Control, Power, Equipment.

the small supply ship. In any case, the resulting costs will be in reasonable proportion to the benefits derived from the existence of a scientific laboratory in space.

Summary and Conclusions

A new orbital supply system is proposed which uses automatically guided, wingless supply ships for cargo transport and restricts the application of upper glider stages to passenger service only.

A comparison of these two configurations, on the basis of about 1.3 million pounds take-off weight, indicates that the ratio of payload weight over gross weight is about nine times higher for the automatic carrier than for a vehicle with winged upper stage. It is shown that actually the design requirements for cargo transport are very different from those governing passenger transport so that, by a separation of both functions, controversial compromises can be avoided, resulting in higher design efficiency, safety, and economy.

From an analysis of orbital supply as well as space flight operations, it is concluded that two types of automatic supply vehicles are needed: one large vehicle with about 11,000-lb payload for periods of increased supply demands, and one small supply ship with about 2000-lb payload for regular service.

A take-off weight of about 1.3 million pounds is obtained for the large supply ship, as well as for the passenger ship. The latter carries a payload of 5 persons, corresponding to about 1200 lb.

As a measure of the supply economy in general, the take-off weight factor, defined as ratio of take-off weight to delivered payload weight, has been introduced. The transport efficiency in particular is described by expressing the propellant consumed and the dead weight expended during each supply flight in terms of delivered payload, called propellant and dead-weight factor, respectively.

By using automatic supply vehicles the take-off weight factor is reduced from 1120 to 122 and the propellant factor from 945 to 105. The dead weight factor is reduced from 158 to 16, assuming that all wingless stages are lost, while the third stage of the passenger vehicle is recovered. In the case of parachute recovery of wingless first stages, the dead-weight factor is reduced to 27 and 3.45, respectively, for passenger and supply ship.

The 2000-lb supply ship weighs about 275,000 lb, tandem-staged. Its efficiency is comparable to that of the large supply ship. Accepting a higher take-off weight, further economic improvements are possible by using a winged first stage as carrier, or by parachute recovery of stage 1 and use of a winged second stage for glide recovery where parachute recovery becomes doubtful. To provide both booster stages with wings would increase the take-off weight very much. In any case, the use of a small supply ship reduces the level of effort required for maintenance. This is very important, because maintenance operations must be conducted throughout the lifetime of the orbital installation.

Since the small ship with tandem arrangement is well within the capability of rocket technology a few years from now, the important development tasks are the large automatic supply ship, hypersonic rocket gliders for booster stages, and the orbital glider. Essential for these projects is the introduction of rocket engines, operating with more powerful propellants than presently in use, in order to avoid unnecessarily high take-off weights.

The requirements for building a small satellite and equipping a circumlunar expedition, using the large supply ship, have been discussed as a numerical example. In both cases, the efforts are of comparable magnitude requiring about 50 successful supply flights.

The energy required to equip and carry out a given interplanetary expedition can be minimized by proper selection of the orbits of departure and arrival. For the transfer ellipse and for the flight maneuvers indicated in Fig. 4[10], an optimum orbit, requiring minimum energy for the particular transfer ellipse, has been determined analytically. For this orbit, the planetary escape energy is equal to the solar potential energy difference between the two planets. It is also shown, however, that the best assembly orbit (which at the same time is the orbit of departure) lies as close to the earth as possible. The effect of return into the optimum satellite orbit is discussed by example of a small Mars expedition. The expedition needs in this case about 5.8 million pounds of propellant and 1 million pounds of other material, having saved about 870,000 lb of initial take-off weight by returning into the optimum orbit. The resulting reduction in propellant expenses for the supply service amount to about 90,000,000 lb, not to mention other expendable material. This is nearly 12 times the over-all material requirement for the Mars expedition. For the latter, 550 to 600 successful supply flights with the 11,000-lb carrier into the assembly orbit are needed.

This preliminary analysis conducted in this paper shows that separate passenger and material service can be used to improve the economy of orbital and space operations. While research to obtain access to more powerful energy sources is of vital long-range importance, improved engineering methods on the basis of chemical propellants, combined with the observation of practical limits in establishing the objectives, actually seem to hold the key to space, at least for the next few decades.

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[10] For Fig. 4, see Part 1 in the September-October issue of JET PROPULSION, pp. 302-309.

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(*Continued on page 365*)

Thermal Properties and Applications of High-Temperature Aircraft Insulation

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Fundamental principles of heat transfer in fibrous insulations are discussed. Thermal conductivity of a high-temperature insulating felt is plotted against mean temperature at various densities, and the conductivity data are analyzed to determine the contribution by each of the mechanisms of heat transfer to the total conductivity. Radiation and air conduction account for nearly all the conductivity at high temperatures. Conductivity due to radiation varies approximately as the cube of the absolute temperature, linearly with the effective fiber diameter, and inversely as the density of the insulation. Conductivity due to air conduction depends only on temperature and cannot be controlled by varying the physical properties of the insulation. Thermal conductance of a given insulating felt may be reduced by increasing either thickness or density, the former being more efficient from the standpoint of minimum insulation weight. An internal reflective foil does not appreciably reduce the conductance of an insulating felt.

Applications of high-temperature insulation to protect aircraft structures are illustrated in specific heat-transfer problems. Criteria are discussed for evaluating the performance of a jet-engine insulating blanket. Computations are carried out that show the relationship between engine and fuselage temperatures, the conductance of the felt, the emissivity of the enclosing foil, and the air flow rate between the blanket and the fuselage, for fixed ambient conditions. Problems that illustrate thermal function of insulation under transient heat-flow conditions are described and solved with minimum mathematical detail.

Nomenclature

(See Appendix for discussion of thermal insulation terms)

b = actuator radius, ft.
C = thermal conductance of insulation ($Q/\Delta t$), Btu/(hr ft² deg F)
c = specific heat of insulation, Btu/(lb deg F)
D = thermal diffusivity of insulation ($k/12\rho c$), ft²/hr
$\bar{d}$ = mean fiber diameter, in.
$\bar{d^2}$ = mean square fiber diameter, in.²
G = temperature gradient, deg F/in.
H = heat capacity of spar per unit area, Btu/(deg F ft²)
H_a = heat capacity of actuator per ft length, Btu/(deg F ft)
h = thermal conductance of air layer ($Q/\Delta t$), Btu/(hr ft² deg F)
h_1, h_2, h_3, h_4 = convection coefficients, Btu/(hr ft² deg F)
k = thermal conductivity (Q/G), Btu/(hr ft² deg F/in.)
k_a = thermal conductivity due to air conduction, Btu/(hr ft² deg F/in.)
k_r = thermal conductivity due to radiation, Btu/(hr ft² deg F/in.)
L = insulation thickness, in.
p = porosity of insulation
Q = heat flow, Btu/(hr ft²)
R_1 = thermal resistance of insulation ($1/C$), (hr ft² deg F)/Btu
R_2, R_3 = thermal resistance of air layers ($1/h$), (hr ft² deg F) Btu
S = average pore size, in.
T, T_1, T_2, T_a = absolute temperatures, deg R
t, t_1, t_2, t_a = temperatures, deg F
T_f, t_f = air temperature, deg R, deg F
T_0, t_0 = equivalent isothermal temperature of blanket foil, deg R, deg F
T_w, t_w = wall temperature, deg R, deg F
T_x, t_x = actuator temperature, deg R, deg F
t_s = spar temperature, deg F
U = transmittance (see Appendix, Equation [10]), Btu/(hr ft² deg F)
α = effective opacity of insulation fiber
β_n = roots of equation $\beta \tan(\beta L/12) = \rho c/H$
ϵ_2, ϵ_3, ϵ_4 = surface emissivities
ρ = bulk density of insulation, lb/ft³
ρ_0 = true density of fibrous material, lb/ft³
τ = time, hr
Δt = temperature drop across insulation or air layer, deg F

Heat Transfer in Fibrous Insulations

HIGH-temperature aircraft insulation generally consists of a felt enclosed in metallic foils. The felt is composed of high-silica or alumino-silica fibers which do not soften or devitrify at temperatures in excess of 2000 F. Fiber diameters range from one to ten microns, and the felt is highly porous. In nearly all applications the air spaces occupy more than 90 per cent of the total insulation volume. Heat transfer through the felt occurs by air conduction and convection, radiation, and fiber conduction. The purpose of the insulation is to break up a large air space into a great number of small pores, thereby reducing heat transfer by convection and radiation.

Fig. 1 shows the contribution by each of the mechanisms of heat transfer to the thermal conductivity of a felt used for aircraft insulation. Conductivities are plotted against insulation density for temperatures of 500, 1000, and 1500 F.

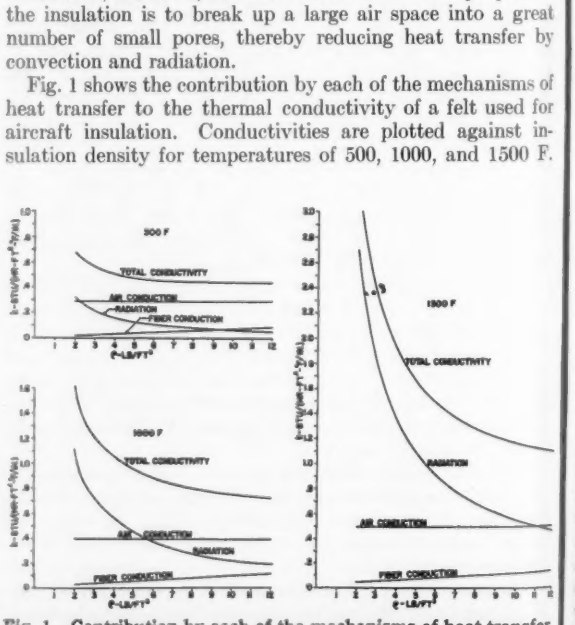


Fig. 1 Contribution by each of the mechanisms of heat transfer to the total conductivity at 500, 1000, and 1500 F

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It is seen that air conduction and radiation are the important mechanisms of heat transfer, the latter becoming dominant as the temperature is raised and the density lowered. The curves for air conduction are based on Keyes's (1)[2] empirical equation for the conductivity of air, and the radiation curves were plotted from Equation [4], which is discussed further in greater detail. Total conductivity curves at 500 and 1000 F were obtained from thermal conductivity measurements with the guarded hot plate (2), while the curve at 1500 F was computed from Equation [1]. Air convection was shown to be negligibly small at 150 F in insulations of practical density (3), and it would be expected to be even more negligible at higher temperatures since the Grashof number for air decreases with increasing temperature. Heat conduction along the tortuous fiber paths is extremely small, as was shown by low-pressure conductivity measurements (3, 4). However, the fibers contribute indirectly to the thermal conductivity since they occupy a small fraction of the insulation volume that does not contribute any appreciable resistance to heat flow. As a good approximation, the thermal conductivity of a fibrous insulation at high temperatures is given by

$$k = \frac{k_a + k_r}{p} + a\rho \quad \ldots\ldots\ldots\ldots\ldots\ldots [1]$$

The porosity in the denominator of the first term, which is generally between 0.9 and 1.0, accounts for most of the small contribution by fiber conduction. The second term, which may be disregarded for most practical computations, takes into account the additional parallel conduction along the tortuous fiber paths. For the insulation upon which Fig. 1 is based, excellent agreement between experimental conductivity values and those predicted by Equation [1] was obtained by setting a equal to 0.005. Both k_a and k_r are temperature-dependent, the former varying roughly as the square root of the absolute temperature and the latter as the cube of the absolute temperature.

Air conduction in an insulating felt cannot be reduced, unless the felt is enclosed in an evacuated space, since the pore size is always appreciably greater than the mean free path of the air molecules, roughly 4×10^{-6} in. Conductivity due to radiation, on the other hand, depends on the density of the felt and the diameters and optical properties of the fibers. Radiation emitted by one fiber passes through an air space and is incident upon a neighboring fiber, where part of the radiation is absorbed and the remainder is scattered in all directions or transmitted through the fiber. Hamaker (5) has developed a theory of heat conduction by radiation within powdered materials. Extension of this theory to fibrous insulations, and assuming "gray"-body radiation, yields for the conductivity due to radiation

$$k_r = 1.386 \times 10^{-8}\, T^3\, (S/\alpha) \quad \ldots\ldots\ldots\ldots [2$$

The effective opacity of each fiber, α, takes into consideration both absorption and scattering. The average pore size of the insulation may be computed from elementary measurements of density and fiber diameter, by the formula (3)

$$S = 0.785\, (\overline{d^2}/\overline{d})\, (\rho_0/\rho) \quad \ldots\ldots\ldots\ldots [3]$$

The factor (S/α) in Equation [2] may be regarded as the mean free path for radiation quanta. If the fibers were "black"-bodies, α would be unity and the free path length would simply be S, the average pore size. Since heat conductivity by radiation is proportional to $(\overline{d^2}/\overline{d})$, the effective fiber diameter, a fibrous insulation used at elevated temperatures should be composed of extremely fine fibers. Scattering of infrared radiation is also dependent on the fiber diameter distribution, so α depends on the fiber diameters as well as on the chemical composition of the fibrous material. For the fibrous insulation upon which Fig. 1 is based, α is 0.50 and the effective fiber diameter is 3.6 micron. For this insulation, the conductivity due to radiation is given by

$$k_r = 0.72 \times 10^{-9}\, T^3/\rho \quad \ldots\ldots\ldots\ldots [4]$$

The numerical constant on the right side of Equation [4] was actually determined empirically from Equation [1] and 55 thermal conductivity measurements over a mean temperature range 500 to 1100 F and at densities of 2 to 12 lb/ft³. The excellent fit of the experimental conductivity values, with k_r assumed proportional to $T^3\rho^{-1}$, justifies the assumption of gray-body radiation over the temperature range 500 to 1100 F. This assumption is implicit in the computed conductivities from 1100 to 1500 F. Fiber opacity α was calculated from Equations [2], [3], and [4] and elementary measurements of fiber density and fiber diameter distribution. Infrared transparency measurements on very thin samples of the felt, using the Perkin-Elmer infrared spectrometer, showed that the transparency was nearly constant over the wavelength range 2 to 9 micron, with the exception of a very narrow transmission band near 7 micron, and the transparency dropped off virtually to zero at longer wave lengths. These infrared measurements further justify the assumption of gray-body radiation, at least for temperatures up to 1000 F.

Since the dependence of k_r on $T^3\rho^{-1}$ is general for fibrous insulations, the density of an insulation has a profound effect on its conductivity at high temperatures. In Fig. 2, the total

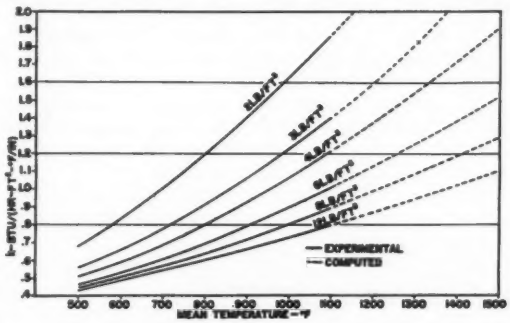


Fig. 2 Thermal conductivity of a high-temperature felt at densities vs. mean temperature

conductivity of the felt considered in Fig. 1 is plotted against mean temperature for various densities. These curves are based upon the 55 thermal conductivity measurements on the felt. In application the temperature of an insulation varies from hot to cold surfaces, and the effective conductivity may be obtained by an integration. For practical calculations, however, the conductivity corresponding to the mean temperature of the insulation is a sufficiently good approximation to the effective conductivity. An insulating felt exposed to 1500 F on the hot side and 500 F on the cold side possesses an effective conductivity that is very nearly equal to its true conductivity at 1000 F.

The thermal conductance of a felt determines its insulating value. Conductance of a given felt may be reduced by increasing its thickness or density. For applications on flat surfaces or where the radius of curvature is large compared with the thickness of the insulation, an increase in thickness produces a proportional reduction of the conductance of the felt. An increase of density reduces only heat transfer by radiation, but not by air conduction. Therefore, increasing the density produces less than a proportional reduction of the conductance of the insulation, considerably less if the temperature is sufficiently low or the density already sufficiently high that radiation is not the dominant mechanism of heat transfer. The weight of insulating felt required to provide a definite conductance at a specified mean temperature is reduced by increasing thickness at the expense of density, and this is frequently an important consideration in jet propulsion applications.

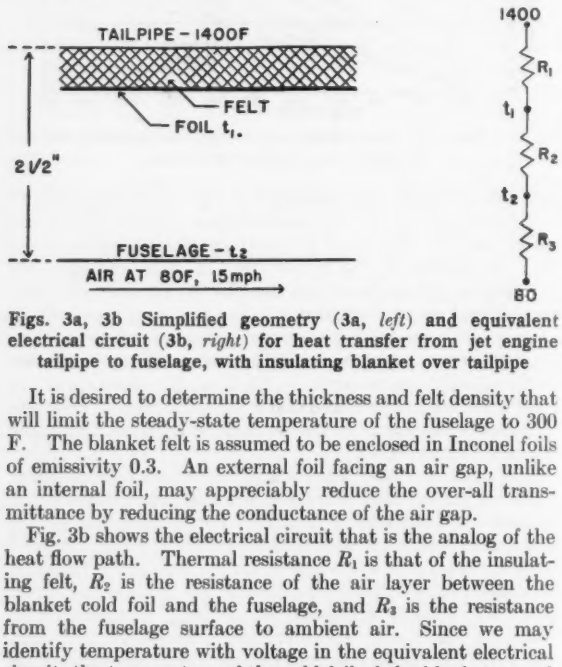
[2] Numbers in parentheses refer to References on page 378.

Reflective foils have been inserted between felts in insulating blankets to reduce thermal conductance. Such internal foils do not appreciably reduce heat transfer by radiation within the insulation, since the effectiveness of an internal foil is limited to an average distance (S/α) on either side of the foil. Computations based on Equation [2] indicate that a highly reflective foil centered between two $^1/_4$-in. thick felts at 4 lb/cu-ft density and 800 F mean temperature would produce only a 2 per cent reduction of conductance. This was verified by thermal conductance measurements on jet-engine insulating blankets with and without center aluminum foils.

Evaluation of Insulation Performance

In aircraft applications the performance of an insulation is evaluated by its ability to reduce the temperature rise of a structural member. A functional test of the insulation is made by measuring either the temperature rise of the structural member under specified operating conditions or the reduction of the cooling load for a given temperature rise. For design purposes, the insulation requirements can be computed from known thermal conductance data. This is illustrated by a simple heat-transfer problem.

Suppose a jet-airplane insulating blanket is fastened around a 3-ft OD tailpipe section at 1400 F. A painted fuselage with emissivity 0.9 is spaced $2^1/_2$ in. from the tailpipe, as shown in Fig. 3a. For simplicity, we first assume still air between the blanket and the fuselage. Ambient air temperature t_a is assumed to be 80 F, with a 15-mph air velocity transverse to the direction of heat flow over the outside of the fuselage. These conditions are similar to those under which the performance of a jet insulating blanket might be evaluated in a test cell or on an airplane on the ground with the engine running.



Figs. 3a, 3b Simplified geometry (3a, *left*) and equivalent electrical circuit (3b, *right*) for heat transfer from jet engine tailpipe to fuselage, with insulating blanket over tailpipe

It is desired to determine the thickness and felt density that will limit the steady-state temperature of the fuselage to 300 F. The blanket felt is assumed to be enclosed in Inconel foils of emissivity 0.3. An external foil facing an air gap, unlike an internal foil, may appreciably reduce the over-all transmittance by reducing the conductance of the air gap.

Fig. 3b shows the electrical circuit that is the analog of the heat flow path. Thermal resistance R_1 is that of the insulating felt, R_2 is the resistance of the air layer between the blanket cold foil and the fuselage, and R_3 is the resistance from the fuselage surface to ambient air. Since we may identify temperature with voltage in the equivalent electrical circuit, the temperature of the cold foil of the blanket t_1 and that of the fuselage t_2, under steady-state conditions, are computed from

$$t_1 = 80 + \frac{1320\,(R_2 + R_3)}{R_1 + R_2 + R_3} \quad \text{[5a]}$$

$$t_2 = 80 + \frac{1320\,R_3}{R_1 + R_2 + R_3} \quad \text{[5b]}$$

The fuselage temperature t_2 is the indicator of insulation performance. Frequently, however, insulation specifications are based on a maximum allowable temperature for the cold foil of the blanket. Equations [5a] and [5b] show that t_1 is not a good indicator of the performance of the insulating blanket. If the thermal resistance R_2 is increased, t_2 is lowered while t_1 is raised. With still air between the blanket and the fuselage, heat transfer through the air layer is due primarily to radiation, and R_2 is extremely dependent on the emissivity of the blanket foil. A highly reflective foil, such as polished aluminum, would contribute to a lower t_2 and a higher t_1, while a badly oxidized stainless-steel foil would increase t_1 and decrease t_2. In general, the temperature attained by the blanket cold foil is a poor criterion for judging the performance of the insulating blanket.

Thermal resistances are computed from the formulas

$$R_1 = C^{-1} \quad \text{[6a]}$$

$$R_2 = \left[6.93 \times 10^{-9}\,\epsilon_2 \left(\frac{T_1 + T_2}{2}\right)^3 + h_2\right]^{-1} \quad \text{[6b]}$$

$$R_3 = \left[6.93 \times 10^{-9}\,\epsilon_3 \left(\frac{T_2 + T_a}{2}\right)^3 + h_3\right]^{-1} \quad \text{[6c]}$$

The first term in each bracket of Equations [6b] and [6c] is the conductance due to radiation,[3] and the second term is the conductance due to forced or free air convection. Emissivity of the fuselage ϵ_3 is 0.9, while the effective emissivity ϵ_2 for radiation between the blanket foil and the fuselage is 0.29. The forced convection coefficient h_3 was computed as 2.90, using the simplified equation of McAdams (6) for turbulent air flow parallel to the axis of a long cylinder. The free convection coefficient h_2, which depends on the unknown temperature t_1, was calculated from curves by Jakob (7) based on experimental studies by Beckmann of free convection in cylindrical gas layers. If we use the limiting value of 300 F for t_2, R_3 is computed from [6c] as 0.21. The values of t_1, R_2, and R_1 are then obtained from simultaneous solution of Equations [5a], [5b], and [6c]. This solution is usually simplified by first estimating a value for t_1, and then using successive approximations to obtain an accurate solution.

For the conditions assumed, the solution yields $t_1 = 720$ F, $R_2 = 0.41$, and $R_1 = 0.64$, and from [6a] the blanket conductance C is 1.56. For a blanket containing the felt upon which Fig. 2 is based, with a 1060 F mean temperature, this conductance is obtained with a $^1/_2$-in.-thick felt at 11 lb/cu ft density or a $^3/_4$-in.-thick felt at 4 lb/cu ft.

It is interesting that, with still air between the blanket and the fuselage, the temperature drop across this air layer is about two thirds that across the blanket. The relatively large value of R_2 under these conditions stresses the importance of the emissivity of the blanket foil. If the emissivity of the outer foil were increased from 0.3 to 0.6, all other conditions remaining unchanged, the fuselage temperature would be increased from 300 to 330 F, while the temperature of the blanket cold foil would actually be lowered from 720 to 630 F.

With appreciable air flow between the blanket and the fuselage, temperatures t_1 and t_2 are computed by setting up two heat-balance equations on the cold foil surface of the blanket and on the fuselage. Suppose that it is desired to estimate the fuselage temperature t_2 that will be realized with an 1800 F tailpipe, a $^1/_2$-in.-thick blanket felt at a density of 6 lb/cu ft, and air flow at 250 F and a mass velocity of one

[3] We obtain a radiation coefficient as a function of the mean temperature from the identity

$$(T_1^4 - T_2^4)/(T_1 - T_2) = 4\left(\frac{T_1 + T_2}{2}\right)^3 + \left(\frac{T_1 + T_2}{2}\right)(T_1 - T_2)^2$$

and note that the second term on the right is less than 10 per cent of the first term if the absolute mean temperature is at least 1.5 times the temperature difference (see Ref. 6, p. 64).

lb/(sec ft²) between the blanket and the fuselage, all other conditions being the same as in the preceding problem. The heat balance equations are

$$C\,(1800 - t_1) = 0.173\,\epsilon_2\left[\left(\frac{T_1}{100}\right)^4 - \left(\frac{T_2}{100}\right)^4\right] + h_1\,(t_1 - 250) \quad [7a]$$

$$0.173\,\epsilon_2\left[\left(\frac{T_1}{100}\right)^4 - \left(\frac{T_2}{100}\right)^4\right] = h_1\,(t_2 - 250) + h_3\,(t_2 - 80) + 0.173\,\epsilon_3\left[\left(\frac{T_2}{100}\right)^4 - 850\right] \quad [7b]$$

where forced convection coefficients h_1 and h_3 are 3.50 and 2.90, respectively, and ϵ_2 and ϵ_3 are 0.29 and 0.90. (Coefficient h_1 was also computed from the simplified formula of McAdams (6) for turbulent flow along a tube, using the equivalent diameter, 4-in., of the annular space.) Since C depends on the mean temperature of the blanket, and therefore on t_1, Equations [7a] and [7b] are most easily solved by successive approximations. The solution yields $t_1 = 730$ F and $t_2 = 265$ F. If the emissivity of the blanket foil is raised from 0.3 to 0.6, the fuselage temperature t_2 rises to 300 F, while the blanket cold foil temperature t_1 drops to 660 F. Again the cold foil temperature does not properly evaluate the performance of the insulating blanket.

Insulation requirements based on steady-state temperatures are on the conservative side. In many aircraft applications, severe conditions are imposed on the insulation for only short intervals of time. Two further examples, taken from actual computations to estimate insulation requirements for aircraft structures, illustrate the function of high-temperature insulations in transient heat-flow applications.

Transient Heat-Flow Applications of Aircraft Insulation

Thermal insulation is required to protect a wing spar, which is a side wall of a fuel tank. The spar, a thin aluminum plate with a heat capacity per square foot of 0.49 Btu/deg F, is adjacent to a burner section of the engine. In the event of a leak, the spar would be exposed instantaneously to 2000 F on its outer surface. It is desired to compute the thickness and density of insulating felt that would prevent the spar temperature from exceeding 250 F for 15 min in such an emergency.

One-dimensional heat flow is assumed for the computations, with a temperature of 2000 F on the hot surface of the insulation established at zero time. The initial temperature of the insulation and spar is 140 F, the ambient temperature inside the fuel tank. To obtain a conservative result, cooling of the spar by the fuel inside the tank is neglected. The thermal diffusivity of the felt, based on a 1000 F mean temperature and a density of 8 lb/cu ft, is 0.033 ft²/hr.

If, as a first approximation, we neglect the cooling of the spar due to convection from its inner surface, the temperature rise may be computed with the equation of Carslaw and Jaeger (8) for a solid in contact with a well-mixed fluid. The solid in this case is the insulating blanket, while the aluminum plate may be assumed to possess the essential property of the well-mixed fluid in that its temperature is constant throughout its thickness, at any instant of time. The Carslaw-Jaeger equation, with the values substituted for initial and hot surface temperatures, yields the following expression for the spar temperature t_s as a function of time τ:

$$\frac{t_s - 140}{1860} = 1 - \sum_{n=1}^{\infty} \frac{2\left(\beta_n^2 + \frac{\rho^2 c^2}{H^2}\right) e^{-\beta_n^2 D\tau} \sin\frac{\beta_n L}{12}}{\beta_n\left[\frac{L}{12}\left(\beta_n^2 + \frac{\rho^2\,c^2}{H^2}\right) + \frac{\rho_c}{H}\right]} \quad [8]$$

The infinite series converges very rapidly, and only the first two terms need be considered. The cooling of the plate due to convection from the inside surface is obtained as a second-order approximation, and this proves to be a very small correction for these conditions.

A computed spar temperature of 260 F was reached after 15 min. with a 2-in.-thick felt, while a 3-in. felt lowered the temperature attained in 15 min. to 170 F. This points out the considerable importance of insulation thickness for transient heat-flow applications where the rate of temperature rise is governed primarily by the rate of heat diffusion through the insulation. For such applications of insulation, the law of time (9) serves as a good approximation: The time required to produce a specified temperature rise varies roughly as the square of the insulation thickness. The insulation should have the lowest possible thermal diffusivity, which is obtained by combining a low conductivity and a high density.

When only a very thin insulation can be used, the heat capacity of the insulation may be neglected, and the rate of temperature rise depends only on the conductance of the insulation and the heat capacity of the insulated structural component. An example of such a problem is the insulation of an exhaust nozzle actuator near a wall at 1300 F, with sufficient space for only a $^1/_2$-in.-thick insulating blanket. The actuator is essentially a 5-in. OD stainless steel cylinder with a heat capacity of 4.56 Btu/deg F per ft length containing a stationary hydraulic fluid with negligible thermal capacity. Air at 700 F flows axially over the outside of the insulation with a mass velocity of one lb/(sec ft²), from which a convection coefficient $h_4 = 2.90$ is computed. The emissivity of the outer blanket foil is assumed to be 0.4, and the density of the insulating felt is 8 lb/ft³. One side of the insulated actuator is exposed to radiation from the 1300 F wall, while the other side receives radiation from a source at a temperature near that of the ambient air, 700 F. The temperature of the hydraulic fluid, initially at 80 F, must not exceed 350 F after 15 mins. of operation under these conditions.

An approximate solution of this problem is obtained by considering the outer foil of the blanket as an equivalent isothermal surface at temperature t_0. Quasi-steady-state heat flow through the blanket is assumed, with a uniform inside surface temperature equal to that of the actuator at any given instant of time. The heat balance on the equivalent isothermal surface and the rate of temperature rise of the actuator are described by the equations

$$0.174\,\epsilon_4\left[\frac{1}{\pi}\left(\frac{T_w}{100}\right)^4 + \frac{1}{2}\left(\frac{T_f}{100}\right) - \left(\frac{T_0}{100}\right)^4\right] = h_4\,(T_0 - T_f) + C\,(T_0 - T_x) \quad [9a]$$

$$H_a\frac{dt_x}{d\tau} = 2\pi b\,C\,(t_0 - t_x) \quad [9b]$$

The solution for T_0 in Equation [9a], with $C = 1.0$,[4] is not highly sensitive to the value of the time dependent temperature T_x, T_0 increasing from 1260 R to 1285 R as the actuator temperature rises from its initial value of 520 R to its maximum allowable value of 810 R. Therefore, we use $T_0 = 1270$ R or $t_0 = 810$ F in Equation [9b]. Solution of [9b] with substitution of appropriate constants yields

$$t_x = 810 - 730\,e^{-0.32\tau} \quad [9c]$$

and the actuator temperature t_x rises only from 80 to 140 F as τ increases from zero to 0.25 hr.

Protection of instruments inside a guided missile against the elevated skin temperatures attained during a brief flight is another possible application of insulation under conditions of transient heat flow. Many other examples could have

[4] Since the blanket thickness is small compared with the actuator radius, the conductance may be computed as that of a $^1/_2$-in.-thick blanket covering a flat surface. The actuator radius b should be taken to the center of the blanket.

been used, covering different temperatures and geometrical configurations, but the two problems selected adequately describe the function of the insulation in retarding diffusion or conduction of heat to the insulated component.

Summary

Experimental conductivity data on a fibrous insulation at various mean temperatures and densities have been analyzed to determine the contribution by each of the mechanisms of heat transfer to the total conductivity. Radiation and air conduction account for nearly all the conductivity at higher temperatures. Conductivity due to radiation, which varies approximately as the cube of the absolute temperature, can be reduced by decreasing the pore size of the insulation, either by increasing the density or by reducing the fiber diameters. Increasing the thickness of a felt produces a proportional reduction in its thermal conductance, while an increase in the density of the felt produces less than a proportional reduction in conductance. An internal reflective foil does not appreciably lower the conductance of an insulating felt. Conductivity data on a high-temperature insulating felt were given as a function of mean temperature and density; data were used for heat-transfer calculations pertaining to jet-propelled aircraft.

High-temperature aircraft insulation generally consists of an alumino-silica or high-silica felt enclosed in metallic foils. If the insulation is used in series with an air gap, the emissivity of the foil may have an appreciable effect on the over-all transmittance. In the application of a jet insulating blanket over a tailpipe, with still air between the cold foil of the blanket and the air frame, the air gap was computed to contribute about 40 per cent of the total thermal resistance from tailpipe to fuselage, with a foil emissivity of 0.3. For such applications of insulation, the performance of the insulation in a functional test should be evaluated from the temperature rise of the structural component being protected, or from the reduction of the cooling load required to limit the temperature rise to a specified value. The temperature attained by the cold foil of the insulating blanket is not a good indicator of the performance of the insulation.

Thermal insulations are frequently used in jet-propelled aircraft to provide protection for structural components against excessive temperature rise under transient operating or emergency conditions. If there is sufficient space for a fairly thick insulation, a low thermal diffusivity of the insulation can provide adequate thermal protection, even though the heat capacity of the insulated component may be very low. If available space permits only a very thin insulation, the heat capacity of the insulation may generally be neglected, and the conductance is the only thermal property of the insulation that is effective in retarding heat transfer. Several problems were described, and solutions given with a minimum of mathematical detail, to illustrate the function of high-temperature aircraft insulations in steady-state and transient heat-flow applications.

APPENDIX

Thermal Insulation Terms

Terms that have common usage in the thermal insulations field are conductivity, conductance, transmittance, and resistance. Equations that define these terms have been given in the nomenclature. Thermal conductivity is a specific property of an insulating material; and the first section of this paper dealt with the relationship between the thermal conductivity of a fibrous insulation and its mean temperature, density, and fiber diameter distribution.

Conductance is the most familiar insulation term used in heat-transfer computations. It has become common practice to distinguish between the conductance of a bulk material and of an air layer or film, although both C and h define the ratio of the heat flow through a medium to the temperature drop across it. For a homogeneous bulk insulation covering a flat or moderately curved surface, the thermal conductance is approximately equal to the conductivity at the mean temperature divided by the thickness of the insulation. The conductance of an air film or layer is frequently denoted as a surface coefficient of heat transfer, which may be broken down into its component radiation and convection coefficients. The term transmittance gives the total conductance of a heat-flow path consisting of two or more conductances in series, such as a bulk insulation in series with an air film. In the example cited, the transmittance is computed from formula

$$1/U = 1/C + 1/h \quad \ldots\ldots\ldots\ldots\ldots\ldots [10]$$

Thermal resistance is a useful concept for solution of heat-flow problems by analogy between a heat-flow path and an electrical circuit. The thermal resistance of a bulk insulation or an air film is the reciprocal of the conductance. The total resistance of a heat-flow path consisting of two or more thermal resistances in series is simply their arithmetic sum; hence, total resistance is reciprocal of the transmittance.

Thermal diffusivity is the important physical property of an insulation for many transient heat-flow applications. It is the ratio of the thermal conductivity to the heat capacity per unit volume of insulation. Since insulation performance is generally evaluated under steady-state conditions, diffusivity is not so familiar a term in the insulations field as those discussed above.

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Transpiration Cooling in Porous Metal Walls

(*Continued from page 368*)

the effective heat-transfer coefficient h_g from measured data on T_g, T_{wc} and T_{wg} in cylindrical walls, independently of losses to the exterior ($h_c \neq 0$) and of the coolant entrance temperature T_{cc}.

Comparison of the plane parallel wall formulas, Equations [16], [17], [18] with those for the cylindrical wall, Equations [16′], [17′], [18′], shows that the latter set may be obtained from the former by use of an equivalent thickness for the cylinder

$$l = r_g \log r_c/r_g \quad \ldots\ldots\ldots\ldots\ldots\ldots [19]$$

with $G_c = G_{cg}$.

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(*Continued on page 386*)

Technical Notes

The Thermal Expansion of Some Liquids of Interest as Rocket Fuels

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The volume temperature expansion coefficient has been determined for nitromethane and three dilutions of hydrazine, over the temperature range 0 to 75 C. Corrections for the expansion of the glass dilatometer were avoided by a preliminary volume calibration with mercury over the complete temperature range.

Introduction

THE significant expansion effect in liquids is the volumetric or bulk expansion. In general, liquids have a greater volume expansion than solids, so that it is needful to know expansion coefficients in order to be able to correctly design liquid containers.

There are two good methods available for measuring liquid expansion coefficients: (*a*) Regnault's method of balancing columns in which the liquid is loaded into a U-tube arrangement, the two sides of which are maintained at different temperatures, and the difference in height measured. This method is independent of the expansion of the apparatus. (*b*) The dilatometer method in which the liquid is expanded in a glass apparatus in which the expansion of the glass is taken into account. If the liquid to be tested is volatile or contains volatile constituents, it is necessary to use a sealed equipment, so that volatile constituents cannot evaporate and change the composition or the quantity of contained liquid. Either of the two methods mentioned in (*a*) or (*b*) may be used for volatile liquids if the containers are sealed.

Because of the simplicity of the equipment, the ease of filling, and the ease of data taking, the method making use of the dilatometer was chosen. The closed dilatometer was first suggested by De Luc in (1).[2] In the form of the equipment used in this experiment, the dilatometer was made of Pyrex glass, with an elongated bulb, the volume ratio of which to the capillary tube was about seven. The amount of liquid was adjusted so that the meniscus extended into the capillary tube at the lowest temperature, which was zero degrees centigrade in these experiments. The liquid was frozen while sealing off the top of the tube so that after it was melted, it expanded against a small pressure.

Apparatus and Calibration Procedure

Fig. 1 is a sketch of one of the glass dilatometers after filling with the liquid and sealing. It is designed with a comparatively long bulb in order to easily withstand an internal pressure of approximately four atmospheres which is developed at the higher temperatures. The length is such that it may readily be weighed by hanging on the beam hook of an ordinary analytical balance.

A method of calibrating and using these tubes which avoids having to make use of the coefficient of expansion of glass is as follows: Mercury is filled into the bulb and extends up into the capillary. The net weight of mercury is determined from the total weight, less the weight of the glass. The tube is then immersed in a temperature-controlling bath, and the height of rise of the mercury meniscus is marked on the glass stem at a number of different temperature points covering the range for which the temperature expansion coefficients are desired. From the known values of mercury densities at different temperatures, it is then possible to make an absolute volume calibration of the glass dilatometer, and also from the same data one can get the volume constant of the capillary tube. It is then easy to get the absolute volume at any point on the capillary by taking the volume at a given temperature as determined from the mercury calibration and adding or subtracting the stem volume correction due to the position of the liquid meniscus at a different point on the stem.[3] The actual volume corrections may be made by a simple graphical method as will be explained later.

After the calibration marks have been cut into the glass stem, using a tungsten carbide knife edge, and the marks checked, the mercury is removed and the dilatometer is filled with the liquid to a point such that the meniscus will be in the capillary tube at the lowest temperature point of 0 C. For hydrazine, the filling must be done in a nitrogen dry-box. For any liquid, the filling must be done in such a way that there is no loss or change of composition of the liquid. For filling, it is found convenient to use a long pointed pipette, thin enough to reach into the dilatometer bulb. This may be operated with a hypodermic syringe inside a dry-box. The correct amount of liquid having been placed in the dilatometer tube, it is then sealed off. During the sealing-off process, the liquid may be cooled or frozen in dry ice.

The sealed-off dilatometer may be used like an ordinary thermometer; that is, one may place it in a bath of known temperature and mark the height of the meniscus on the capillary stem. Using this procedure, the position of the meniscus was noted and marked on the capillary tube for six different points up to 75 C. The container was a Dewar flask of sufficient height to almost completely immerse the tube. Temperatures were measured with thermometers which were estimated to 0.1 C.

The distances between the marks on the glass capillary were measured with a Cenco traveling microscope equipped with a centimeter scale and vernier reading to 0.001 cm. A single traverse was made for both the mercury and the fuel expan-

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[2] Numbers in parentheses refer to References at end of paper.

[3] Another way of avoiding the correction for the glass expansion is to add an amount of mercury to the dilatometer, the expansion of which is just equal at each temperature point to the glass expansion. This is the so-called constant volume dilatometer. For example, see "Textbook of Heat" by Stewart, Satterly, and Temperly, University Tutorial Press, London, 1950, p. 67.



Fig. 1 Sketch of the glass dilatometer filled and sealed. Calibration marks are shown as they appear after marking with a tungsten-carbide knife edge. The mercury and fuel marks were measured in a single traverse of the traveling microscope

Although this figure is placed horizontally, the dilatometer is always used in a vertical position.

EDITOR'S NOTE: This section of JET PROPULSION is open to short manuscripts describing new developments or offering comments on papers previously published. Such manuscripts are published without editorial review, usually within two months of the date of receipt. Requirements as to style are the same as for regular contributions (see first page of this issue).

sion points. This permitted the mercury and fuel expansion data to be measured on the same linear scale so that the stem correction was obtained by a simple graphical method which is explained below. The effect of linear expansion of the glass tube was neglected, since the maximum was of the order of 0.005 cm, or less than the uncertainty of positioning the marks on the glass.

Data and Calculations

Experiments were made with a number of hydrazine-water mixtures and with dehydrated and purified nitrobenzine. In each case the liquid was sealed into a calibrated dilatometer and tested at six temperature points between 0 and 75 C. A permanent record was made by marking the meniscus positions on the dilatometer stem, and measurements of these marks with a traveling microscope gave data which made possible the determination of the volume temperature relations.

An example of the recording and reducing of the results will now be given. Fig. 2 is a set of curves plotted for nitromethane. In this figure, curve *A* gives the absolute volume of the dilatometer bulb and tube up to the mercury meniscus at the given temperature. These volumes were determined from the weight density ratio for the contained mercury using published density values for mercury (2). Curve *B* is the plot on the linear scale of the mercury calibration points, and curve *C* is the plot on the same linear scale of the nitromethane expansion points.

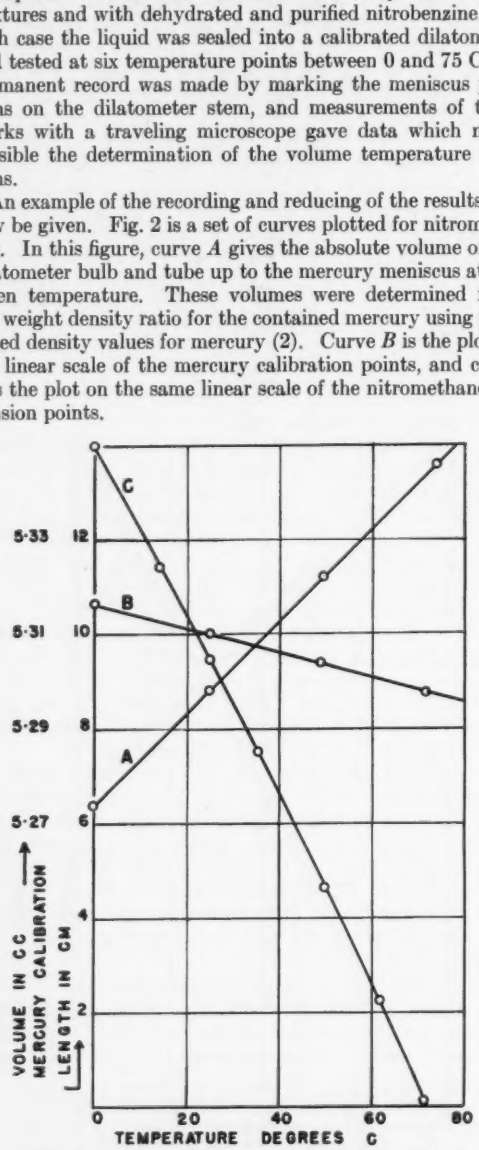


Fig. 2 Graphs of data for a glass dilatometer filled with purified nitromethane

A = volume temperature data (cc scale); B = mercury temperature date (cm scale); C = nitromethane temperature data (cm scale)

It will be seen, then, that the difference in ordinates of curves *B* and *C* at any given temperature represents a length of the tube of the dilatometer. The volume of this differential tube length which is the amount of liquid above the calibrated volume point must be added to the known volume (curve *A*) to obtain the volume of the fuel at any given temperature.

The process will be illustrated by two points computed from Fig. 2. Taking 5.274, the volume of the dilatometer from curve *A* at 0 C. and subtracting 3.3., the difference between curves *B* and *C*, multiplied by 0.0377 cc, the volume of unit length of stem, we get 5.1497, the corrected volume of the nitromethane at 0 C. Likewise, taking 5.3130, the volume of the dilatometer from curve *A* at 40 C and adding 2.99 cm, the difference between curves *B* and *C* multiplied by 0.0377 cc per cm, we get 5.4257, the volume of the nitromethane at 40 C.

We are now able to compute the temperature coefficient of expansion from the defining equation which is

$$\beta = \frac{V_2 - V_1}{V_1(t_2 - t_1)} \quad \ldots\ldots [1]$$

where V_2 is the volume at a temperature t_2 and V_1 is the volume at t_1. Using the above volumes and temperatures, we get, for β between 0 and 40 C

$$\beta = \frac{5.4257 - 5.1497}{5.1497\ (40\ \text{C})}$$
$$= 1.34 \times 10^{-3}\ \text{cc per cc per C}$$

If the value of β is computed as above over the temperature range of 0 to 75 C, we get $\beta = 1.42 \times 10^{-3}$. Actually, as will be seen by inspection of the curve *C*, the data are not linear but require some higher-order terms for complete representation. The volume over the complete range may be expressed in the form

$$V_t = V_0(1 + at + bt^2 + ct^3) \quad \ldots\ldots [2]$$

and from this, β may be deduced at any temperature by expressing [1] above in the form

$$\beta = \frac{1}{V_0}\frac{dV}{dt} \quad \ldots\ldots [3]$$

A set of curves similar to Fig. 2 was plotted for each liquid. From these curves the corrected volumes at temperatures of 0, 20, 40, 60, and 75 C were worked out. These results are tabulated in Table 1.

The data for each liquid given in Table 1 was reduced to the form of Equation [2]. The coefficients, and also the value

Table 1 Volume Temperature Data for Liquids Between 0 and 75 C

	0 C	20 C	40 C	60 C	75 C
Nitromethane, 99.9%	5.1497	5.2853	5.4257	5.5774	5.6980
Hydrazine, 82.9%	5.1915	5.2366	5.2864	5.3403	5.3853
Hydrazine, 93.9%	5.1463	5.2433	5.3409	5.4454	5.5265
Hydrazine, 98.5%	5.0268	5.1146	5.2071	5.3063	5.3864

Table 2 Cubical Expansion Coefficients for Some Rocket Fuel Liquids and Liquid Mixtures. The Data Have Been Fitted to the Equation $V_t = V_0(1 + at + bt^2 + ct^3)$

	$\beta \times 10^3$ at 25 C	$a \times 10^3$	$b \times 10^6$	$c \times 10^9$
Nitromethane, CH_3NO_2 99.9%	1.427	1.29302	0.69944	1.32026
Hydrazine, N_2H_4, 82.9%	0.4871	0.42033	0.72579	0.40714
Hydrazine, N_2H_4, 93.9%	0.9361	0.86661	0.55358	0.55769
Hydrazine, N_2H_4, 98.5%	0.9538	0.85706	0.64608	0.85869

of β at 25 C are given in Table 2. The nitromethane was a purified sample prepared in this laboratory from a commercial product. The estimate of 99.9 per cent is based on the fact that the density checked closely and the index of refraction checked exactly with recently published data (3). The three samples of hydrazine were hydrazine-water mixtures, and the weight percentage of hydrazine was determined in each case by the direct iodate method (4).

Acknowledgments

The author wishes to express his appreciation to Martha Kenyon for the data reduction resulting in the constants of Table 2, and also wishes to express his gratitude to Dr. Howard Kruse for the supply of hydrazine used in this experiment and to Charles Drew for the purified sample of nitromethane.

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Satellite Orbits for Interplanetary Flight

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IN A recent publication (1),[2] the energy equation for entering an elliptic orbit around the sun was differentiated to find the most economical satellite orbit of departure or arrival. The general nature of the analysis did not require a specification of the type of transfer operation (number of impulses or burning periods of the space vehicle). However, the partial derivative Equation [14] in (1) was evaluated on the basis of the one-impulse method, as indicated in Fig. 4 of (1). The resulting orbital distance was designated as optimum distance, $r_{s,\,opt}$.

D. F. Lawden (2) has shown that the two-impulse method (hyperbola-ellipse-circle, or vice versa, as illustrated in Fig. 1 of this note) can be more economical than the one-impulse method. Lawden found that with this method the transfer energy requirement becomes progressively smaller with decreasing ratio r_P/r_A (or b/a in his nomenclature) of the intermediate ellipse. The condition for the superiority of the two-impulse method, Lawden pointed out, is that the residual velocity after escape from the planet (here called hyperbolic excess) is greater than the parabolic (or escape) velocity from the respective satellite orbit of departure or arrival. Conversely, if the hyperbolic excess is smaller than the local parabolic velocity, the one-impulse method is more economical.

The minimum condition Equation [16a] from (1) represents exactly the point at which both methods are equal (hyperbolic excess = local parabolic velocity). At the same time, Equation [15] in (1) shows that at this point the one-impulse method itself has a minimum. The term $r_{s,\,opt}$ therefore applies only to the one-impulse method. Thus, with increasing r_s, the two-impulse method shows a continuously decreasing energy requirement, while the one-impulse method first decreases considerably faster (when $r_s < r_{s,\,opt}$), while later on it indicates an increasing energy requirement (for $r_s > r_{s,opt}$). Consequently, it is correct to state that, unless orbital distances $r_s > r_{s,\,opt}$ can be used, $r_{s,\,opt}$ represents the minimum energy distance available with either method.

Fig. 6 in (1) shows that the minima for the one-impulse method are a function of the solar distance of the target planet (or target apsis). For Venus and Mars the minima are very far out, and they are broad, so that r_s can be reduced considerably below $r_{s,\,opt}$, in fact, by as much as 10,000 to 20,000 n.mi., without appreciably increasing the one-impulse energy requirement. Fig. 1 compares both methods for the case of Mars as target planet. It can be seen that, for a comparable reduction in r_s, the two-impulse method would yield a significantly higher energy requirement. As a corollary to the characteristics of the two-impulse method, it may be noted that in the region $r_s < r_{s,\,opt}$, this method is more economical if $r_P/r_A = b/a$ is not as small as possible.

A numerical evaluation of Equation [15] in (1) shows that in most cases orbital distances which are likely to be of greatest interest for various reasons, lie in a region where the one-impulse method will apply. For Mars as target planet, the two-impulse method, in order to be more economical, requires $r_s > 49{,}000$ n.mi. This is undesirable for departure (1) and also for arrival (3). For Venus as target planet, $r_{s,\,opt}$ lies even farther out. The conclusions obtained in (1) and (3) regarding orbits of departure and arrival from Venus and Mars remain therefore correct. However, for arrival from Jupiter or from trans-Jovian planets, the two-impulse method can be more economical within the same range of orbital distances which has been allowed for arrival from Venus and Mars.

With respect to operations near target planets, one finds, for flights from and to the earth

Planet	$\gamma(10^3$ n.mi.^{3}/sec$^2)$	$r_{s,\,opt}$ (n.mi.)	$y_{s,\,opt}$ (n.mi.)
Mercury	2.94	208	−1,184
Venus	50.80	48,100	44,700
Mars	6.64	3,525	1,695
Jupiter	19,710.0	4,265,000	4,425,000
Saturn	5,560.0	1,291,000	1,260,000
Uranus	885.0	280,000	268,000
Neptune	1,030.0	433,000	418,000
Pluto	5.89	3,000	0

Again, with the exception of Mercury and Pluto, one finds that for most of the interesting regions, $r_s < r_{s,\,opt}$. The Martian moons lie outside this region; however, for practical reasons, a closer distance might be desirable, especially in case

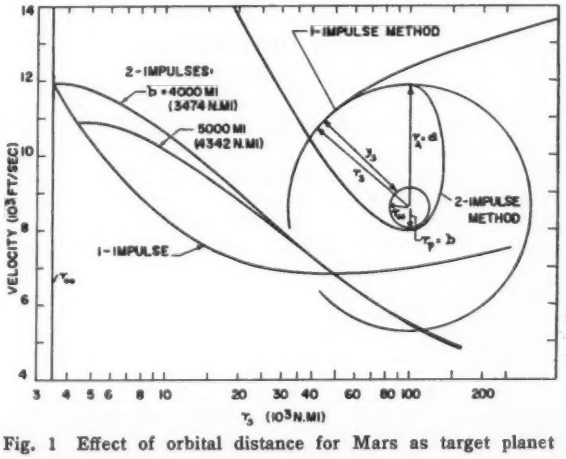


Fig. 1 Effect of orbital distance for Mars as target planet

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[2] Numbers in parentheses refer to References at end of the paper.

of a landing. The same holds true for Venus. The one-impulse method applies also to Jupiter's interesting large satellites, Io, Europa, Ganymede, and Callisto, as well as to Saturn's Titan.

It may finally be remarked that the partial derivative Equation [14] in (1) actually contains a second minimum solution, besides the one leading to $r_{s,\,opt}$, namely, $\gamma/2r_s^2 = 0$; hence, either $\gamma = v_s^2 r = 0$ (i.e., $r_P = 0$), or $r_s = r_A = \infty$. This solution has no significance for the one-impulse method, but it can be interpreted as representing Lawden's minimum for the two-impulse method, $b/a = r_P/r_A = 0$.

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3 "Analysis of Orbital Systems," by Krafft A. Ehricke. Paper presented at the 5th Congress of the International Astronautical Federation at Innsbruck, Austria, August 1954.

Comment on "Satellite Orbits for Interplanetary Flight"

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Mr. Ehricke's note is most helpful in further elucidating the relationship between the two modes of escape from, or entry into, a satellite orbit. However, his Fig. 1 indicates that the position as regards optimum satellite orbits is essentially simpler than that which he has described in part I of his paper, "A New Supply System for Satellite Orbits" (1). If we assume that the most economical mode of escape or entry is always selected (i.e., employing one or two impulses), his graph shows that the ideal velocity requirement for the maneuver steadily decreases as the radius of the orbit increases, there being a change-over from one to two impulses as r_s increases through $r_{s,\,opt}$. We may therefore say simply that the radius of the optimum satellite orbit should be as large as it can conveniently be made after all other requirements have been taken into consideration. From this point of view, the significance of the orbital radius $r_{s,\,opt}$ is not that it represents an optimum condition, but that it marks the boundary between the one- and the two-impulse domain. As Ehricke points out, convenient values of r_s will generally be less than this transition value and, consequently, escape will usually be effected by a one-impulse maneuver.

Response Time in Temperature-Indicating Instruments

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THE effective use of required temperature data in control systems demands a knowledge of the response time of temperature measuring device. The response time of an instrument is arbitrarily defined as the time taken for a thermal measuring instrument to experience 63.2 per cent of an instantaneous temperature change in the medium whose temperature it is designed to measure. For a given temperature measuring device, this response time may vary widely depending upon the conditions of use. For example, the response time will be much lower when measuring stagnant air temperature than it would be in measuring the temperature of a high velocity air stream. In practice it is not generally feasible to attempt to determine either analytically or experimentally the response time under all possible conditions of operation. Furthermore, even if response time were completely known for all conditions of operation, the derivation of actual instantaneous temperature from the indicated temperature information would require rather complex computing devices, thereby increasing the size and complexity of the finished instrument.

In actual practice, in order to avoid the difficulties mentioned above, it becomes highly desirable to develop temperature measuring devices having a sufficiently fast response time so that the temperature indication can be considered essentially identical, moment for moment, with the actual temperature to be measured. Theoretically, this would require an infinitely fast response time for the indicating instrument. Practically, however, certain minimum conditions can be prescribed for the response time such that the difference between indicated temperature and actual temperature may be kept within desirable limits over all expected conditions of operation. These minimum conditions can be described in terms of simple experimental procedures involving step temperature changes.

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Nomenclature

e = base of Napierian logarithms
f = frequency, cps
k = rate of temperature rise coefficient of thermal element, degrees per sec per degree temperature difference
r = recovery factor, dimensionless. Also, summation index
t = time unit, sec
t_r = response time, sec
T = temperature of thermal element
T_s = temperature of medium whose temperature is being measured
T_0 = initial temperature of thermal element
ΔT = $T_s - T_0$
T_t = stagnation temperature of moving air stream
T_m = measured temperature of moving air stream
T_f = free stream temperature of moving air stream
α = proportionality constant
ω = angular frequency, radians per sec

Analysis

The differential equation governing the conduction of heat from a medium to a thermometer element, when the phenomenon is linear with temperature difference, is given by

$$dT = k(T_s - T)\,dt \quad \ldots\ldots\ldots\ldots\ldots\ldots [1]$$

If T_s is constant, the variables may be separated, and direct integration between the initial temperature T_0 (at $t = 0$) and the temperature T at any time t, yields

$$T = T_s - \Delta T e^{-kt} \quad \ldots\ldots\ldots\ldots\ldots\ldots [2]$$

and the instantaneous error in measurement is

$$T_s - T = \Delta T e^{-kt} \quad \ldots\ldots\ldots\ldots\ldots\ldots [3]$$

The response time of the instrument is arbitrarily defined as the time t_r taken for the instantaneous error to be reduced to $1/e$ of the change ΔT. From Equation [3] the response time is thus related to the coefficient k by the relationship

$$kt_r = 1 \qquad \text{or } t_r = 1/k \quad \ldots\ldots\ldots\ldots\ldots\ldots [4]$$

The relationships [2] and [4] are shown graphically in Fig. 1.

If the temperature T_s is a function of time only, the solution to Equation [1], if we assume the initial temperature T_0 of the probe to be an arbitrary zero reference level, will be

$$T = ke^{-kt} \int_0^t T_s e^{-kt}\,dt \quad \ldots\ldots\ldots\ldots\ldots\ldots [5]$$

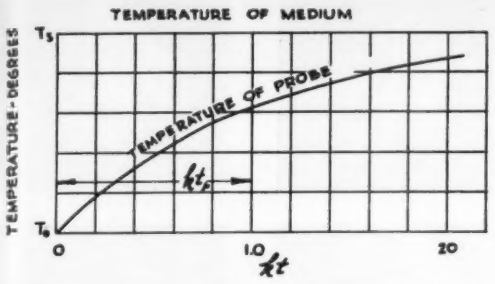


Fig. 1 Temperature-time variation of thermal element in constant temperature medium

Under conditions where the function under the integral sign is not integrable or is known only through an experimentally derived curve, the integration may be conducted graphically.

As a specific instance in the application of Equation [5], let us assume that the function is a linear one and varies in accordance with the expression $T_s = \alpha T$. The solution to Equation [5] in this instance is given by

$$T = \alpha t - (\alpha/k)(1 - e^{-kt}) = T_s - \alpha t_r(1 - e^{-kt}) \ldots [6]$$

and the instantaneous error by

$$T_s - T = \alpha t_r(1 - e^{-kt}) \ldots [7]$$

This solution is plotted in Fig. 2 along with the assumed variation of the temperature of the medium. It is seen that after the initial transient interval the temperature of the thermal element will lag the temperature of the medium by an amount αt_r and that this amount is directly and simply related to the response time of the instrument. In this instance it may be readily seen that the maximum permissible lag called for in specifications may be simply related to the required response time to meet these specifications.

Some interesting and useful deductions may be made by expressing the temperature variation of the medium (the zero temperature level being the initial temperature of the probe) as an infinite Fourier series in accordance with the expression

$$T_s = \sum_{r=0}^{r=\infty} A_r \sin(r\omega t) + B_r \cos(r\omega t) \ldots [8]$$

Substitution of [8] into [5], and term-by-term integration, yields

$$T = \sum_{r=0}^{\infty} \frac{1}{1 + (r\omega/k)^2} \times \{A_r(\sin r\omega t - r\omega/k \cos r\omega t) + B_r(\cos r\omega t + r\omega/k \sin r\omega t)\} + e^{-kt} \sum_{r=0}^{\infty} \frac{1}{1 + (r\omega/k)^2} \left(\frac{r\omega}{k} A_r - B_r\right) \ldots [9]$$

The terms under the first summation constitute the steady-state solution, while the terms under the second summation sign constitute the transient solution, and by appropriate manipulation the error, after the initial transient, will be

$$T_s - T = \sum_{r=0}^{\infty} \frac{1}{1 + (k/r\omega)^2} \times \left\{A_r\left(\sin r\omega t + \frac{k}{r\omega} \cos r\omega t\right) + B_r\left(\cos r\omega t - \frac{k}{r\omega} \sin r\omega t\right)\right\} \ldots [10]$$

Examination of Equations [9] and [10] show that in order to indicate accurately any particular frequency component r, the factor

$$\frac{1}{1 + (r\omega/k)^2}$$

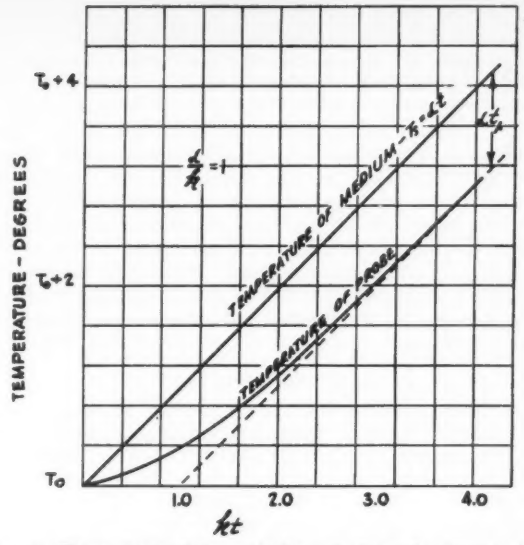


Fig. 2 Temperature-time variation of thermal element in medium of linearly rising temperature

must be near unity, i.e., $r\omega/k \ll 1$, the degree depending on the accuracy required. This condition may be alternately expressed as $2\pi f t_r \ll 1$.

If a criterion of maximum amplitude error permissible for the highest specified frequency component, f, is, say 1 per cent, this implies that the response time must be less than about $^1/_{60}$ of the time taken for a whole cycle at this highest specified frequency.

In instances where the expected temperature variation of the medium is known approximately through experimental information, it may not be necessary or desirable to express the variation as a Fourier series and, in fact, attention need only be focused on the most severe portion of the variation. Equation [5] may be integrated directly, using integration by parts. A single integration yields

$$T = T_s - e^{-kt} \int_0^t e^{kt} \frac{dT_s}{dt} dt \ldots [11]$$

The second term on the right gives the instantaneous error, and this may be calculated by graphical integration if the expected variation in T_s is known or can be estimated.

Continued differentiation by parts will yield

$$T_s - T = \sum_{r=1}^{n} (-1)^{r+1} \frac{1}{k^r} \left\{\left(\frac{d^r T_s}{dt^r}\right)_t - e^{-kt}\left(\frac{d^r T_s}{dt^r}\right)_0\right\} + (-1)^n \frac{e^{-kt}}{k^n} \int_0^t e^{kt} \frac{d^{n+1} T_s}{dt^{n+1}} dt \ldots [12]$$

where the nomenclature $(\ )_t$ and $(\ )_0$ indicates the value of the indicated quantity at time t and at time zero, respectively.

In the event that [12] converges, and in practical cases it generally will, it may be used to compute the expected error $T_s - T$ by plotting graphically the expected variation of T_s and its time derivatives, in the region of the particular moment t of interest and at time $t = 0$. Further, if the system has been in operation a sufficient time so that the transient term under the summation sign can be neglected, then an estimate of the error can be made from observations on the function T_s and its derivatives in the region of the time of interest. This method may be useful in the event that only one severe variation of the temperature T_s may be expected (such as starting of a rocket motor).

Conclusions

The establishment of satisfactory design criteria for the measurement of temperatures, where the permissible error is known, can be directly related to the response time of the instrument, which response time can be determined very simply under experimentally controlled conditions. The availability of accurate high-speed recorders for the determination of response times of a small fraction of a second simplifies the procedure considerably. Furthermore, the determination of this response time will permit the designer to predict the behavior of the instrument under an extremely wide variety of conditions. This is very important from the designer's viewpoint, inasmuch as it is generally difficult or impossible to try to simulate actual operating conditions which will be met.

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Liquid Jet Atomization by a Sonic Nitrogen Stream

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AS PART of a research project on a rocket thrust chamber, a study was undertaken on the atomization and penetration of liquid (water) streams injected at various velocities and angles into a sonic gas stream.

The apparatus (Fig. 1) consisted of a plenum chamber with a tank pressure of 35 psi, exhausting through a 0.250-in. orifice, yielding a well-defined sonic stream of nitrogen.

Water at controlled injection pressures was obtained by pressurizing a 40-gal storage tank. Flow was controlled by a solenoid valve in the delivery line. A hand valve isolated the injection pressure gage from dynamic surges encountered at shutdown.

An adjustable rack mount indexed the orifice under test at any desired impingement angle. The reference plane for the angle of impingement consisted of the axis of the gas jet.

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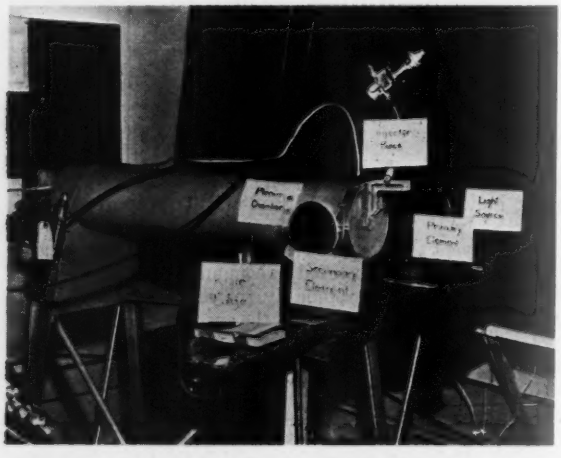
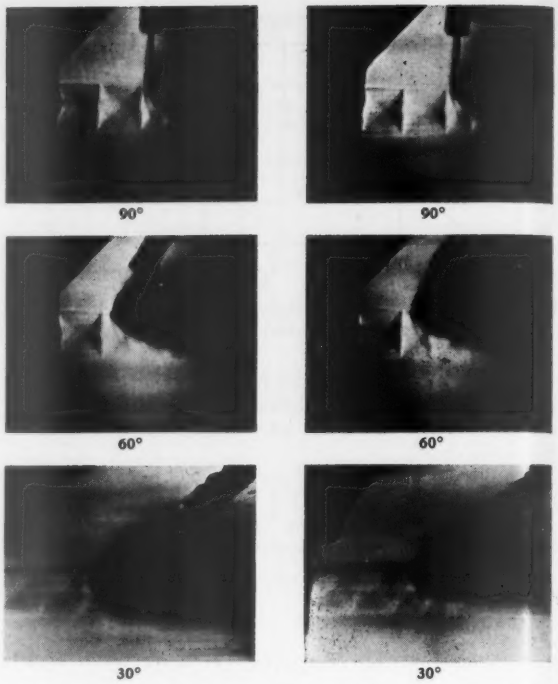


Fig. 1 Plenum chamber, sonic nozzle, injector, and schlieren assembly



ΔP = 150 psi; V_{inj} = 97 in./sec ΔP = 100 psi; V_{inj} = 79 in./sec

Fig. 2 Penetration of 0.020-in.-diam water jet at 30°, 60°, 90°

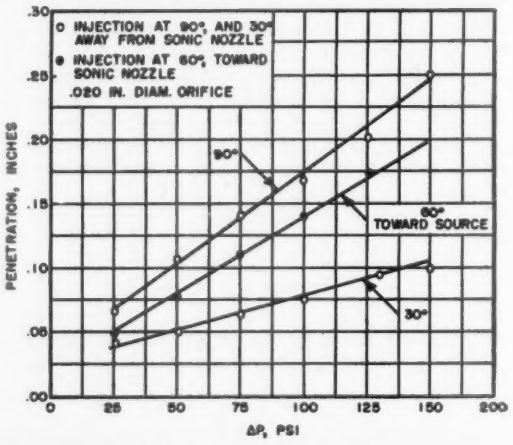


Fig. 3 Effect of injection pressure in liquid jet penetration

A simple double-lens collimated beam schlieren optical train was used in the tests, with a 100-watt Zirconium arc point source and a $^3/_{32}$-in. diam circular knife edge. All measurements of liquid stream penetration and dispersion were made visually on a calibrated Fresnel screen. Since the Zirconium arc source was continuous, short duration flash photographs were not possible. The photographs shown were taken at $f/4.5$ with a $^1/_{1,000}$-sec focal plane shutter; they are illustrative only and were not used for measurements.

The sample curves show the distance of penetration of a 0.020-in. diam water jet, injected at 90°, 30° downstream, and 60° upstream, toward the sonic nozzle. Figs. 2, 3.

Although the penetration was greatest at the 90° impingement angle, the rate or amount of atomization was greater at the upstream impingement angles and seemed greatest at approximately 60°. Both liquid jet penetration and atomization were decreased with downstream impingement (away from the sonic nozzle) until at 10° the liquid stream no longer penetrated the sonic jet.

The total penetration in all cases was a linear function of the liquid jet velocity. No appreciable interaction occurred between the sonic and liquid jets until actual contact was made at the envelope of the sonic jet.

The Influence of Duct Losses on Jet Propulsion Devices

C. A. GONGWER[1]

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Introduction

THE general expression for propulsive efficiency of a jet device such as a ducted propeller, turbojet, or other such system which derives its thrust by accelerating the surrounding medium, is

$$\eta = \frac{2}{2 + (\Delta v/v)} \quad \ldots\ldots [1]$$

where Δv is the absolute wake velocity. This is the velocity increase given to the medium by the jet engines. It is equal to the relative exit jet velocity minus the forward speed. v is the forward speed.

This equation is based on the one-dimensional flow theory in which the velocity profile of the exit jet at the nozzle is rectilinear and without appreciable transverse components. This assumption is sufficiently accurate for this analysis. Equation [1] indicates that as $\Delta v/v$ approaches zero, the value of η increases to unity. It is also evident that as $\Delta v/v$ is allowed to become smaller, the amount of the fluid medium passed through the jet engine must be increased to maintain the thrust. In view of these two facts, although it appears desirable to go as far as possible in the direction of decreasing $\Delta v/v$ to obtain high propulsive efficiency, there is a practical lower limit below which the over-all efficiency of the system is lowered. This is because the losses from ducting, diffusion, etc., attendant upon handling larger quantities of the fluid medium, begin to increase rapidly as $\Delta v/v$ becomes too small. Since very high values of $\Delta v/v$ are also associated with lowered efficiencies, it is apparent that there is an optimum region of $\Delta v/v$ for each type of jet engine.

Derivation of the Optimum $\Delta v/v$

The nature of this optimum $\Delta v/v$ will now be derived. It is assumed that a nose entrance is used or that the boundary layer is dumped in the case of a side entrance so that all the fluid medium entering has the same relative total head $v^2/2g$. The losses associated with handling the fluid medium consist largely of diffuser losses, interference losses, losses due to additional external skin area made necessary by the presence of the jet engine, and duct losses. These losses can be added and the total related to the energy of the entering free stream, $v^2/2g$, by the loss factor K. In other words, the effective velocity head of the entering fluid is reduced from the free-stream value by a factor K so that the effective inlet head can be expressed as

$$H_i = \frac{v^2}{2g}(1 - K) \quad \ldots\ldots [2]$$

The head of the exit jet is

$$H_e = \frac{(v + \Delta v)^2}{2g} \quad \ldots\ldots [3]$$

Therefore the effective energy added per pound of fluid by the actuator inside the jet engine is given by [3] − [2]

$$\Delta H = \frac{2\Delta v v + \Delta v^2}{2g} + K\frac{v^2}{2g} \quad \ldots\ldots [4]$$

where ΔH represents the input of energy in foot pounds per pound of the fluid medium necessary to produce a given $\Delta v/v$ with the losses $K(v^2/2g)$ present.

The propulsive work per pound of the fluid medium handled is given by the impulse per pound, $\Delta v/g$, times the forward speed, or

$$W_p = \frac{\Delta v v}{g} \quad \ldots\ldots [5]$$

The effective jet propulsive efficiency as seen by the actuator inside the duct, be it a pump, fan, or turbo engine, is then [5] divided by [4] or

$$\eta_e = \frac{2}{2 + (\Delta v/v) + K(v/\Delta v)} \quad \ldots\ldots [6]$$

The similarity between [6] and [1] is apparent. Equation [6] is plotted in Fig. 1 as a family of curves for different values

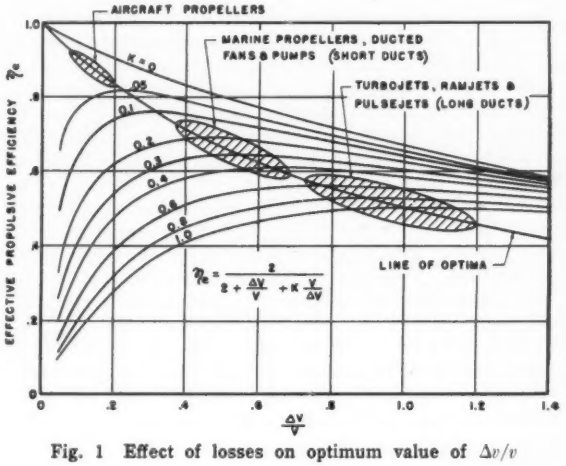


Fig. 1 Effect of losses on optimum value of $\Delta v/v$

of K. Each curve of constant K has an optimum and the locus of these optima is obtained in the usual manner from [6].

$$\frac{\delta\eta_e}{\delta(\Delta v/v)} = 0 = \frac{-2[1 - K(v/\Delta v)^2]}{[2 + (\Delta v/v) + K(v/\Delta v)]^2} \text{ or } K = \left(\frac{\Delta v}{v}\right)^2 \quad \ldots\ldots [7]$$

This line is also shown on Fig. 1. The optimum values of $\Delta v/v$ are seen to increase rapidly with increasing values of K, the loss factor.

Analysis of the Loss Factor K

Some of the separate losses making up K are given by

$$K = K_D + K_{L/D} + K_s + K_i \quad \ldots\ldots [8]$$

Where

K_D = inlet diffuser loss factor
$K_{L/D}$ = ducting loss
K_s = loss due to additional external skin area made necessary by the presence of the jet engine
K_i = loss due to interference of the exit jet with the flow around the propelled craft

All other losses inside the duct may be considered in the performance of the actuator or engine.

K_D is computed from diffuser tests and may lie between 0.40 and 0.10, depending upon the Mach number and the percentage of external diffusion.

$K_{L/D}$ for a long straight entrance duct of length L and diameter D is dependent on the value of L/D and the fraction, M, of the free-stream velocity head remaining in the duct.

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[1] Manager, Underwater Engine Division.

$$K_{L/D} = 0.02 \frac{L}{D} \times M \quad \text{.....[9]}$$

where 0.02 is the straight pipe friction factor. Any serious bends or elbows would increase the loss $K_{L/D}$ markedly.

K_s, the loss from the additional external skin area made necessary because of the presence of the duct, is computed by standard methods and referred to the inlet velocity head by means of Equation [1] as a factor, since the external skin loss must be made up by jet power. If a nacelle is used on supporting struts, this loss may be large.

K_i, the jet interference loss, is appreciable only if the jet is located elsewhere than at the tail end of the body.

Discussion

Since the losses making up K are not greatly affected by the scale of an engine, provided it is always similar in its different sizes, a given engine travels along a line of constant K in Fig. 1 as its size, and hence $\Delta v/v$ is changed for a given thrust requirement. Consequently, if an engine's operating point, plotted on Fig. 1, lies to the left of the line of optima, its performance can be improved by reducing its size and hence increasing the $\Delta v/v$ until it falls on the line of optima. Conversely, if the operating point lies to the right of the line of optima, the engine is too small for the thrust required of it.

The additional practical consideration of the physical size of the jet engine may dictate even larger values of $\Delta v/v$ than are given by the above considerations, since the larger $\Delta v/v$ becomes the smaller is the engine for the same thrust. The optima are rather flat at $\Delta v/v$ around unity and the efficiency may not suffer too much by substantial reductions in size. These considerations are separate from the scope of this paper, however, wherein the only concern is maximum effective efficiency. This general discussion and the curve, Fig. 1, apply to propellers also. The operating points of aircraft and marine propellers and jet engines are indicated on Fig. 1, and it is seen that they fall on or near the line of optima. The range of $\Delta v/v$ in each type can be thought of as function of the blade aspect ratio in the case of propellers, and the length to diameter of the duct in the case of jet engines.

Performance of Ethylene Oxide as a Monopropellant[1]

IRVIN GLASSMAN[2] and JOHN E. SCOTT, JR.[3]

Princeton University, Princeton, N. J.

IN A recent technical note reporting the theoretical performance of ethylene oxide as a monopropellant, Robison (1)[4] considered two possible exothermic decomposition reactions. One reaction leads to the formation of CO and CH_4, and the other to CO, H_2, and C. The first reaction gives the highest adiabatic flame temperature for the decomposition of ethylene oxide, but this reaction is not realized in practical rocket motors. The second reaction gives abnormally low results and is probably unrealistic for processes occurring in these units.

Since it is very unlikely, at the temperatures observed, that CO and CH_4 are the sole products of the decomposition process, calculations were carried out in this laboratory assuming formation of CO, CH_4, CO_2, O_2, H_2, C_2H_4, and H_2O. Using heats of formation, enthalpies, and equilibrium constants from National Bureau of Standards Circular no. 461, it was found that the number of moles of CO_2, O_2, and H_2O formed were very small and could be neglected. Consequently, it is suggested for rocket performance comparisons that the theoretical decomposition reaction of ethylene oxide be chosen as

$$C_2H_4O(l.) \rightarrow a\ CO + b\ CH_4 + c\ H_2 + d\ C_2H_4$$

Table 1

Performance of Ethylene Oxide as a Function of Pressure

P_c, Atm	Moles CO a	Moles CH_4 b	Moles H_2 c	Moles C_2H_4 d
20	1.00	0.84	0.16	0.08
30	1.00	0.86	0.14	0.07
40	1.00	0.87	0.13	0.07
60	1.00	0.88	0.12	0.06

(Continued)

T_c, °K	$\overline{M.\ W.}$	$\bar{\gamma}$	c^*, m./sec	Isp, sec
1288	21.15	1.174	1189	159.4
1300	21.26	1.173	1194	168.0
1306	21.32	1.172	1196	173.6
1312	21.36	1.171	1197	180.7

The results of these calculations for various chamber pressures and injection of the liquid monopropellant at 298 K are presented in Table 1. (The heat of formation of liquid ethylene oxide at 298 K was taken as −18.29 Kcal/mole. (The heat of formation of the gas is reported by the National Bureau of Standards (2) as −12.19 Kcal/mole, and the heat of vaporization is reported by Giauque and Gordon (3) as −6.1 Kcal/mole.) In Table 1, T_c is the adiabatic temperature of the decomposition process, c^* the characteristic exhaust velocity, P_c the chamber pressure, $\bar{\gamma}$ the ratio of mean specific heats of the gas mixture, $\overline{M.\ W.}$ the mean molecular weight of the gas mixture, and Isp the specific impulse calculated, assuming frozen equilibrium and 1 atmosphere exhaust pressure. The products of decomposition are for one mole of reactant. It is seen that the decomposition reaction is not very sensitive to pressure.

For comparison with Robison's results (1) for the decomposition reaction going to CO and CH_4 only, a value of Isp of 166.3 sec, which corresponds to T_c = 1430 K and γ = 1.161, was found for a pressure ratio of 20.42:1.

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Received September 15, 1954.

[1] The calculations reported in this paper were carried out in connection with research being performed under Contract PR-N6ori-105, Task Order III, for Project SQUID.

[2] Research Associate, Department of Aeronautical Engineering. Mem. ARS.

[3] Research Assistant, Department of Aeronautical Engineering. Stud. Mem. ARS.

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Transpiration Cooling in Porous Metal Walls

(Continued from page 378)

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Index of Papers Published in JET PROPULSION, Journal of the American Rocket Society

Volume 24

January-February to November-December 1954

A-K

ABRAMSON, B. N., BRANDWEIN, D. S., and MENES, H. C.
The 350,000-Pound Thrust Rocket Test Stand at Lake Denmark, N. J. 291

ADAMSON, T. C., JR., *see* Marble, F. E., and Adamson, T. C., Jr.

BARTAS, J. G., *see* Mayer, E., and Bartas, J. G.

BEIGHLEY, C. M., and DEAN, L. E.
Study of Heat Transfer to JP-4 Jet Fuel 180

BLATZ, P. H.
Kinematics of a Vertical Booster 37

BLEYLE, G. A., HINCKLEY, R. B., and JEWETT, C. L.
An Air-Transportable Liquid Oxygen Generator—Its Operation and Application 297

BRANDWEIN, N. S., *see* Abramson, B. N., Brandwein, D. S., and Menes, H. C.

CAMPAGNA, F., and KREITH, F.
Comments on Supersonic Ramjet Diffusers 253

CASH, H. S.
A Recoverable Propulsion System for Long-Range Rockets 322

CHARVONIA, D. A., *see* Wood, H. L., and Charvonia, D. A.

CHENG, SIN-I
High-Frequency Combustion Instability in Solid Propellant Rockets. Parts 1 and 2 ..... 27, 102

CLAUSER, F. H.
Ramjet Diffusers at Supersonic Speeds ... 79

CROCCO, G. A.
The Crucial Problem in Astronautics: Recovery of Multistage Vehicles 313

DEAN, L. E., *see* Beighley, C. M., and Dean, L. E.

DEMETRIADES, S. T.
On the Decompression of a Punctured Pressurized Cabin in Vacuum Flight .. 35

EHRICKE, K. A.
A New Supply System for Satellite Orbits—Parts 1 and 2 ..... 303, 369
Satellite Orbits for Interplanetary Flight 381

ESCH, R., *see* Gross, R. A., and Esch, R.

FOSTER, C. R., *see* Summerfield, M., Foster, C. R., and Swan, W. C.

FOSTER, R. W., *see* Stehling, Kurt R., and Foster, R. W.

FRIEDMAN, R.
High-Temperature Durability of Molybdenum in Oxygen-Deficient Combustion Gases 187

GECKLER, R. D., and SPRENGER, D. F.
The Correlation of Interior Ballistic Data for Solid Propellants 22

GLASSMAN, I., and SCOTT, J. E., JR.,
Performance of Ethylene Oxide as a Monopropellant 386

GOLDSMITH, S., and PENNER, S. S.
On the Burning of Single Drops of Fuel in an Oxidizing Atmosphere 245

GONGWER, C. A.
The Influence of Duct Losses on Jet Propulsion Devices 385

GRAHAM, R. W., *see* Reese, B. A., and Graham, R. W.

GREEBLER, P.
Thermal Properties and Applications of High Temperature Aircraft Insulations 374

Issue	Page Numbers
January-February	1– 68
March-April	69–144
May-June	145–212
July-August	213–280
September-October	281–348
November-December	349–420

GREEN, L., JR.
Erosive Burning of Some Composite Solid Components 9
Unstable Burning of Solid Propellants 252

GROSS, R. A., and ESCH, R.
Low-Speed Combustion Aerodynamics ... 95

HINCKLEY, R. B., *see* Bleyle, G. A., Hinckley, R. B., and Jewett, C. L.

JEWETT, C. L., *see* Bleyle, G. A., Hinckley, R. B., and Jewett, C. L.

KAEPPELER, H. J.
Problem of Cooling Nuclear Working Fluid Rockets Operating at Extreme Temperature 316

KEBELY, V., *see* Summerfield, M., Reiter, S. H., Kebely, V., and Mascolo, R. W.

KNUTH, E.
The Mechanics of Film Cooling—Part 1 .. 359

KOVITZ, A. A., *see* Scott, J. E., Jr., Matthews, G. B., and Kovitz, A. A.

KREITH, F., *see* Campagna, F., and Kreith, F.

KRETSCHMAR, G. G.
The Isothermal Possibilities of Some Rocket Propellant Liquids, and the Ratios of the Two Specific Heats 175
Thermal Expansion of Some Liquids of Interest as Rocket Fuels 379

KROP, S.
The Toxicity and Health Hazards of Rocket Propellants 223

L-W

LAWDEN, D. F.
Comment on "Satellite Orbits for Interplanetary Flight" 382

LEES, L.
Fluid-Mechanical Aspects of Flame Stabilization 234

LORELL, J., and WISE, H.
Excitation of Oscillations by Chemical Reaction 110

MACLEOD, G. J.
Production of Chemicals by Gas-Dynamical Methods 111

MANILDI, J. F.
Response Time in Temperature-Indicating Instruments 382

MARBLE, F. E., and ADAMSON, T. C., JR.
Ignition and Combustion in a Laminar Mixing Zone 85

MARSH, B. W., and SEARS, G. A.
Introduction to the Analysis of Supersonic Ramjet Power Plants 155

MASCOLO, R. W., *see* Summerfield, M., Reiter, S. H., Kebely, V., and Mascolo, R. W.

MATTHEWS, G. B., *see* Scott, J. E., Jr., Matthews, G. B., and Kovitz, A. A.

MAYER, E., and BARTAS, J. G.
Transpiration Cooling in Porous Metal Walls 366

MCCARTHY, J.
Nuclear Reactors for Rockets 36

MENES, H. C., *see* Abramson, B. N., Brandwein, D. S., and Menes, H. C.

MIESSE, C. C.
Ballistics of an Evaporating Droplet 237

MINHAS, F.
Comment on "Methods of Flow Measurement" 37

PENNER, S. S., *see* Goldsmith, M., and Penner, S. S.

PRICE, E. W.
Charge Geometry and Ballistic Parameters for Solid Propellant Rocket Motors 16

REESE, B. A., and GRAHAM, R. W.
Heat Transfer and Frictional Pressure Drop Characteristics of White Fuming Nitric Acid 228

REITER, S. H., *see* Summerfield, M., Reiter, S. H., Kebely, V., and Mascolo, R. W.

ROBISON, W. C.
Ethylene Oxide as a Monopropellant 111
Experimental Measurement of the Heats of Dissociation of Hydrazine-Water and Hydrazine-Methyl Alcohol Systems ... 112

SABERSKY, R. H.
Effect of Wave Propagation in Feed Lines on Low-Frequency Rocket Instability. 172

SCOTT, J. E., JR., *see* Glassman, I., and Scott, J. E., Jr.

——, MATTHEWS, G. B., and KOVITZ, A. A.
The 8th ARS National Convention: A Technical Summary 120

SEARS, G. A., *see* Marsh, B. W., and Sears, G. A.

SPRENGER, D. F., *see* Geckler, R. D., and Sprenger, D. F.

STEHLING, K. R., and FOSTER, R. W.
Liquid Jet Atomization by a Sonic Nitrogen System 384

STERLING, G. E.
Utilization of Radio Frequencies in Connection With Rockets 322

SUMMERFIELD, M., FOSTER, C. R., and SWAN, W. C.
Flow Separation in Overexpanded Supersonic Exhaust Nozzles 319

——, REITER, S. H., KEBELY, V., and MASCOLO, R. W.
The Physical Structure of Turbulent Flames 254

SWAN, W. C., *see* Summerfield, M., Foster, C. R., and Swan, W. C.

THOMAS, L. H.
The Vulnerability of Satellite Vehicles to Countermeasures 321

WALTER, H.
Experience With the Application of Hydrogen Peroxide for Production of Power 166

WARES, G. W.
Terminology and Nature of Atmospheric Shells 33

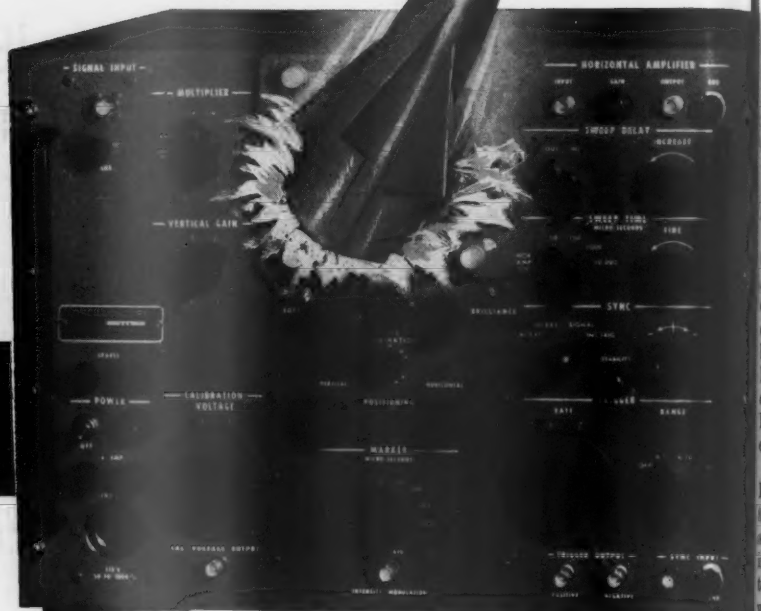
WISE, H., *see* Lorell, J., and Wise, H.

WOOD, H. L., and CHARVONIA, D. A.
The Ignition of Fuel Droplets Descending Through an Oxidizing Atmosphere 162

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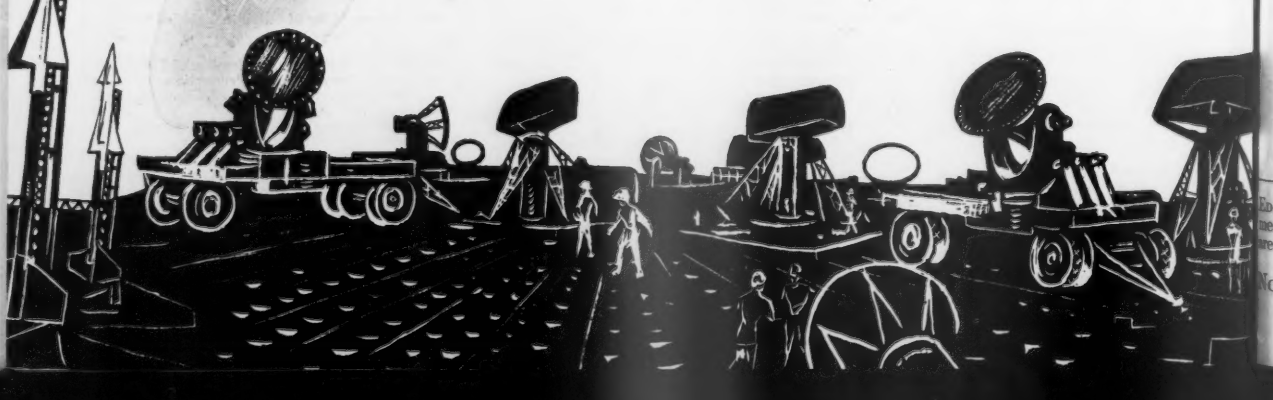
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Jet Propulsion News

Alfred J. Zaehringer, American Rocket Company, *Associate Editor*

Jet Propulsion Progress

BOMBERS. The Air Force officially revealed that a $100 million a year program is being initiated to provide a supersonic bomber. To be built by Convair and known as the XB-58 Hustler, the bomber incorporates a delta-wing design. Power for Hustler is said to come from high thrust turbojets such as the GE J-79 engine of about 15,000-lb thrust.

The official retirement of the B-29 and end of production of the B-36 means increasing emphasis will be given to two of America's leading jet bombers—the B-47 and the B-52. For example, an Air Force B-47 with J-47 turbojet engines has completed 600 flight hours without engine overhaul, equivalent to flying 15 times around the world at the bomber's speed. Previously the Air Force announced a 100% increase in operating time between overhauls which meant that each B-47 could effect a saving of two engines per year. Cost of each engine is about $45,000.

Other savings are yet to be realized, however, since a B-47's RATO take-off costs about $13,000 for 18 RATO's operating for 15 sec. Perhaps for this reason, Aerojet recently demonstrated its liquid propellant RATO for use as an integral installation on the B-47. The B-47 is also proving to be a test vehicle for flight testing of the P&WA J-57 turbojet which will be used on the B-52 and many of our advanced supersonic aircraft. The J-57 is rated at about 10,000-lb thrust.

The first production B-52 recently made its maiden flight. The B-52A differs from its hand-built prototypes in featuring side-by-side cockpit positions and is 156 ft long, has a span of 185 ft, a tail height of 48 ft, and grosses 350,000 lb.

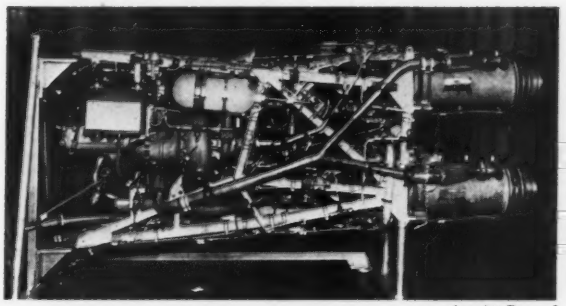
Three new speedy jet bombers appearing on the scene are the Navy Douglas A4D, the Martin B-57, and the Douglas RB-66. The A4D, powered by a Wright J-65 Sapphire turbojet engine of 7200-lb thrust, is a bantam-class bomber and is capable of carrying atom bombs, rockets, machine guns, missiles, or other weapons to suit a wide variety of attack situations. Called the Skyhawk, the A4D is so small that it has been designed without the folding wings traditional in Navy carrier-based planes. Range and performance figures were not given.

Improvements in the Martin B-57 light bomber also reveal it to be a heavily armed aircraft. Featured are a new tandem seating arrangement, fuselage brakes, napalm tanks, and 5-in. HVAR rocket installation. The bomber's engines will be started by a new General Electric unit employing a solid rocket propellant cartridge. The exhaust gases turn a small turbine wheel that is engaged through a clutch and thus starts the main turbine. The RB-66 is a swift, swept-wing, twin-jet reconnaissance bomber and has successfully completed its test flights in California.



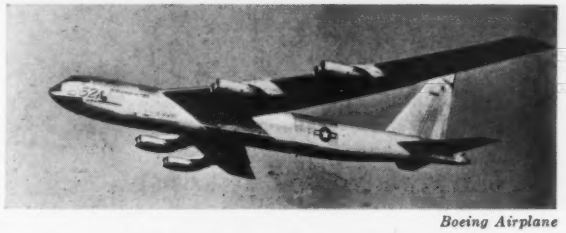
Douglas Aircraft

A4D Bantam jet bomber



Aerojet-General

B-47 liquid propellant ATO rocket power plant



Boeing Airplane

Production B-52A on maiden flight



Glenn L. Martin

Martin B-57B bomber



Douglas Aircraft

RB-66 bomber

EDITOR'S NOTE: The information reported in this Section has been selected from approved news releases originating with the Department of Defense, private manufacturers, universities, etc., and from published news accounts in journal and newspapers. The reports are considered generally reliable, although no attempt has been made to verify them in detail.

FIGHTERS. Several new high-performance fighters such as the F-100 Super Sabre, the Convair delta-wing F-102, and the Grumman F9F-9 Tiber are beginning to roll off the assembly lines. North American has received a $100 million order for production of supersonic F-100 jet fighters. The planes will be manufactured at the North American plant in Columbus, Ohio. Power plant of the craft is the J-57, as is the F-102 all-weather interceptor now going through extensive flight tests. The F9F-9, powered by the J-65, has sonic capabilities in level flight and has a "coke bottle" fuselage which is necked down at the wing roots for drag suppression at high Mach speeds.



Convair

Convair F-102

However, current production models of standard fighters are bristling with more armament. The Navy Chance Vought Cutlass F7U-3 jet fighter has a new fuselage rocket pack which carries 2.75-in. "Mighty Mouse" air-to-air rockets. Although the aluminum belly pack cannot be dropped after firing, it can be reloaded after flight. The Northrop F-89D Scorpion is now equipped with 104 such rockets and is now guarding our frontiers at such advanced bases as Fairbanks, Alaska, just ten minutes jet flying time from the Arctic Circle.



Northrop Aircraft

Scorpion F-89D shows its rocket stingers

Research, even on older jet models, however, is continuing. The Navy Bureau of Aeronautics has developed a system which enables jet aircraft to carry heavier loads and land at slower speeds by means of boundary layer control. The new system bleeds air from the engine and forces it over the trailing edge of the wing, thus reducing turbulence when the plane is at slow or near stalling speeds. In tests at Patuxent River, Md., the new method increased the load capacity of a F9F test plane by 3000 lb and reduced the normal landing speed by 20 knots.

Meanwhile, two new developments are proving of interest on the F-84. In one installation, NACA has demonstrated its new thrust reverser. The device consists of a set of vanes—much like curved venetian blinds which open like a clamshell in the tailpipe. The gases exit through vents in the side of the tailpipe to provide as much as 60% reverse thrust. It weighs about 250 lb now but it is expected that the weight can be cut to 150–200 lb in future models. A loss of about 2–3% forward power results when the blades are not used. In another installation, the USAF has disclosed a radical new fighter, the XF-84H. The plane is an experimental turboprop fighter which is equipped with an afterburner for additional thrust.

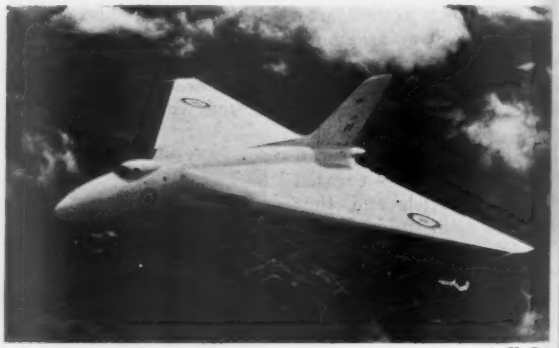
TRANSPORTS. The Boeing 707, a sleek four-jet transport, became America's first jet transport to take to the sky. The 550-mph craft is powered by P&WA JT3-L turbojets of 10,000-lb thrust each. The Air Force is to get 88 of the 190,000-lb ships to serve as high-speed jet tankers. Meanwhile, the Douglas DC-8 jet transport, a four-jet, swept-wing plane capable of nonstop transatlantic flight is fast approaching the hardware stage, while the status of Lockheed's jet transport, the L-193, apparently is awaiting the selection of specific jet engines to become stabilized.



Boeing Airplane

Boeing 707 in the air

However, in Britain, the prototype of the deHavilland Comet 3 has made its maiden flight. Four Rolls-Royce Avon turbojets of 10,000-lb thrust each power the craft which can carry up to 76 passengers, at altitudes of 38,000 to 42,000 ft, to distances of 2600 miles. Other British jet transports are expected to evolve from such military bombers as the delta-winged Avro Vulcan and the crescent-winged Handley Page Victor.



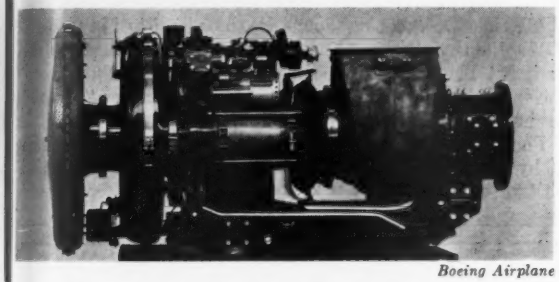
A. V. Roe

Avro Vulcan

France's first jet transport, the SNCASE SE 210 Caravelle is soon scheduled to make its maiden flight. The Caravelle features two jet engines in nacelles on either side of the fuselage just forward of the tail. This location will reduce cabin noise and will allow the swept wings to be built with optimum aerodynamic characteristics. The Hurel-Dubois HD-45 jet transport, also French, apparently is still in the drawing-board stage.

JET ENGINES. There has been considerable activity in developing a light-weight gas turbine engine for automotive purposes. To date there have been four such small engines developed in the United States. These are by Boeing Airplane Company, Chrysler Corporation, General Electric Company, and the General Motors Corporation. A fifth company, the Continental Engine and Aviation Corporation has been actively developing small gas turbines for aviation purposes. However, only the Boeing, Chrysler, and General Motors jet engines are known to have been placed in an automotive vehicle. Several other small turbojets are being used for accessory drives, such as the Solar Aircraft Company "Mars" 50-hp gas turbine engine used as a starter for turbojet aircraft.

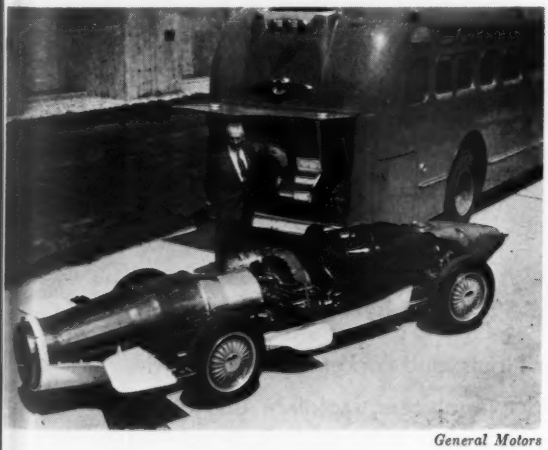
Boeing's engine, the Model 502-10, produces a maximum of 270 hp and has a normal rating of 240 hp. The engine weighs 245 lb dry (320 lb with accessories such as lube system and engine mounts) and can operate on almost any type of fuel without modification. Present installation requires about 15 cu ft. Fuel consumption for the engine (which operates at 2900–3100 rpm) is about 1 lb/hp/hr. The engines have experienced some 30,000 hr of operation, which includes 60,000 mi logged by the first gas-turbine-powered truck, and more than 100 hr powering the Kaman K-25 jet helicopter. Over 300 engines have been built at the Seattle, Wash., plant.



Boeing Airplane

Model 502-10 gas turbine engine

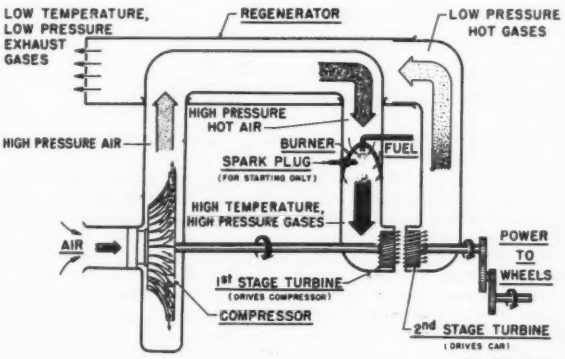
Early in 1953 General Motors began its Firebird Project intended to gain experience in the operation of a gas turbine engine for an automobile. In the Firebird GT-302 Whirlfire



General Motors

Firebird and Turbocruiser

engine, air is drawn into two burners, mixed with fuel, and ignited. After initial ignition, combustion is continuous. The hot exhaust gases turn a turbine which drives the intake compressor. This section is known as the gasifier. The hot gases leave the gasifier and turn the power turbine which is not connected to the gasifier turbine; the two turbines rotate in opposite directions and therefore torque is lessened. The power turbine is connected by a shaft with the transmission gears. Gases from the power section are exhausted directly to the atmosphere. The GM gas turbine engine has had over 1000 hr of laboratory dynamometer tests and over 2500 mi. of experimental driving in the Firebird and in a bus mount—the Turbocruiser. Specifications of the GT-302 include: weight (including transmission and final drive): 775 lb; max. power: 370 hp; gasifier turbine speed (at idle): 8000 rpm; at maximum: 26,000 rpm; operating temp.: 1500 F; startup time is 20 sec.



Chrysler Corp.

Diagram of Chrysler gas turbine

After World War II, Chrysler began work on a regenerative gas turbine. In this system, exhaust gases heat incoming air, thus lowering the final exhaust gas temperature and increasing fuel economy. Early this year the company unveiled its turbine in a conventional automobile. The engine is rated at 120 shaft hp, but can deliver power to the wheels comparable to a 160-hp piston engine. Together with reduction gears, the engine is only 32 in. long, 33 in. wide, and 28 in. high, and is 200 lb lighter and has less than one-fifth as many major moving parts as a piston engine of similar power. In recent road tests, the Chrysler gas turbine installed in a standard production Plymouth averaged 14.9-mpg fuel consumption at a highway speed of 40 mph. The engine has an air compressor unit, a first-stage turbine which drives the compressor and furnishes power for the accessories; the regenerator which recovers heat from the exhaust gas to warm up incoming fresh air; the burner or combustion chamber; and a second-stage turbine, which transmits power to the differential and rear wheels by means of reduction gears.

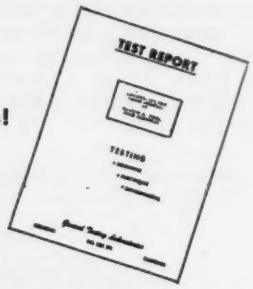
The General Electric light-weight gas turbine engine is called the XT-58. Only details released indicate that it is comparable in size to a conventional automobile but having 6 to 8 times the power. The engine is slated for helicopter applications.

The advantages of present automotive gas turbines are: simple construction, ability to operate on many fuels, high power-to-weight ratio, and a far greater horsepower potential than the conventional piston engine. Present disadvantages, such as high fuel consumption, high noise levels, and high operating temperatures, are gradually being brought into line with the piston engine. Some optimistic engineers are even venturing to predict that the gas turbine powered automobile will be rolling off Detroit assembly lines in ten years.

Three foreign jet engine developments have also proved of interest in the aviation field. Aerojet-General recently demonstrated its thrust reverser, developed from the French SNECMA unit. Providing up to 50% reverse thrust, the unit retracts when not in use and causes no loss in normal positive thrust. A new Rolls-Royce turbojet engine is the 1810-lb thrust "Soar." Measuring 15.75-in. diam and 62.75 in. long, the engine tips the scale at only 267 lb. Meanwhile, a powerful French turbojet has completed successful static

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Aerojet-Genera

Aerojet-SNECMA thrust reverser

tests. This is the Vulcan engine (by SNECMA) which develops 12,000-lb thrust dry and weighs 3100 lb.

Upper Atmosphere Research

PROJECT Skyhook balloons are to be launched in Canada by the U. S. Office of Naval Research. Purpose of the experimental launchings is to collect data regarding the form of the energy spectrum of particles near the low energy "cut-off" for heavy particle primary radiation.

Photographs of the total solar eclipse were made on June 30 by a Navy jet plane in the Minneapolis, Minn., area at an altitude of about 50,000 ft where the atmosphere is relatively

SCINTILLA DIVISION

BENDIX AVIATION CORPORATION

Sidney, New York

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clean and free from haze, cloud formations, dust, and water vapor. The McDonnell F2H-2P photographic jet was armed with a K-38 camera of 36-in. focal length and two movie cameras. Meanwhile, the Office of Naval Research in cooperation with Winzen Research, Inc., also photographed the solar eclipse at altitudes of 40,000–70,000 ft from a Skyhook balloon.

Two new altitude records were set recently. A USAF Air Weather Service balloon reached an altitude of 26 miles in a recent test at Holloman AFB, N. Mex. Previous runs, for charting upper-air conditions for guided missile research, have averaged over 22 miles. A Navy Skyhook balloon set a 117,000-ft mark while carrying 50 lb of instruments for cosmic ray investigations.

A new high-altitude record for manned flight was believed set recently over Edwards AFB, Calif., when the Bell X-1A rocket plane was piloted by Major Arthur Murray to the 90,000-ft level.

Facilities

FOUR new direction finding stations located in Dominican Republic, Puerto Rico, Santa Lucia, and Ascension Island being placed in operation for use with the USAF Missile Test Center at Cocoa, Fla.

North American Aviation is constructing a $4.5 million blow-down wind tunnel near the Los Angeles International Airport. The tunnel will be capable of testing at speeds slower, equal, and faster than sound. The working section measures 17 ft long and has a 4-ft square area. Completion is expected in 1956. Also slated for construction in this area is a new $1.75 million office and laboratory for the Ramo-Wooldridge Corp., electronics and guided missile firm.

Two new facilities are being established on the East Coast. Pratt and Whitney Aircraft will expand to a new atomic-energy aircraft engine facility at South Windsor, Conn., as part of its Air Force and Atomic Energy Commission program. The Republic Aviation Corp. has purchased the modern 424,000-sq ft building formerly owned and operated by the Fairchild Engine Division at Farmingdale, N. Y. The new unit will provide space for engineering operations and is also to be used to consolidate development work now being carried out elsewhere.

Instruments and Materials

STUDIES of combustion, corona-discharge, explosions, plastic and elastic deformation, and shock-wave phenomena can be made with the Beckman & Whitley Framing Camera which can take 25 pictures at framing frequencies up to 2.4 million per sec.

The Oerlikon Tool & Arms Corp. of America has developed a new electronic-optical missile tracking system capable of an accuracy of 0.1 mil at ranges of 3000–100,000 ft.

A chemical milling process has been developed by North American Aviation for precision milling of aircraft and guided missile parts without machining. The chemical method removes unwanted metal from complex or fragile parts at a fraction of the cost of conventional milling. The portion to be removed is exposed while the remainder is masked. The entire part is submerged in an etching solution. Most of the present work has been with aluminum but work is progressing for techniques for use on steel, stainless steel, titanium, and other metals.

The Yardley Electric Corp. has introduced a silver-zinc battery, the Silvercel, to the jet aircraft field. Capable of starting a turbojet ten times on less than one half of its capacity, the new battery has a unit output five times greater than conventional storage batteries. It is already finding wide usage in guided missiles.

PROJECT ENGINEER

for

Coordination of Solid Propellant Development

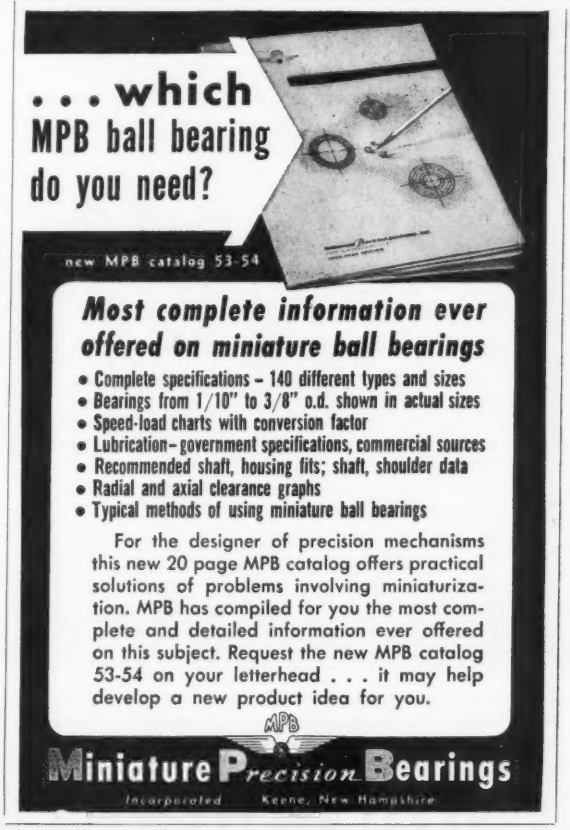
An attractive opportunity for the man who can qualify:

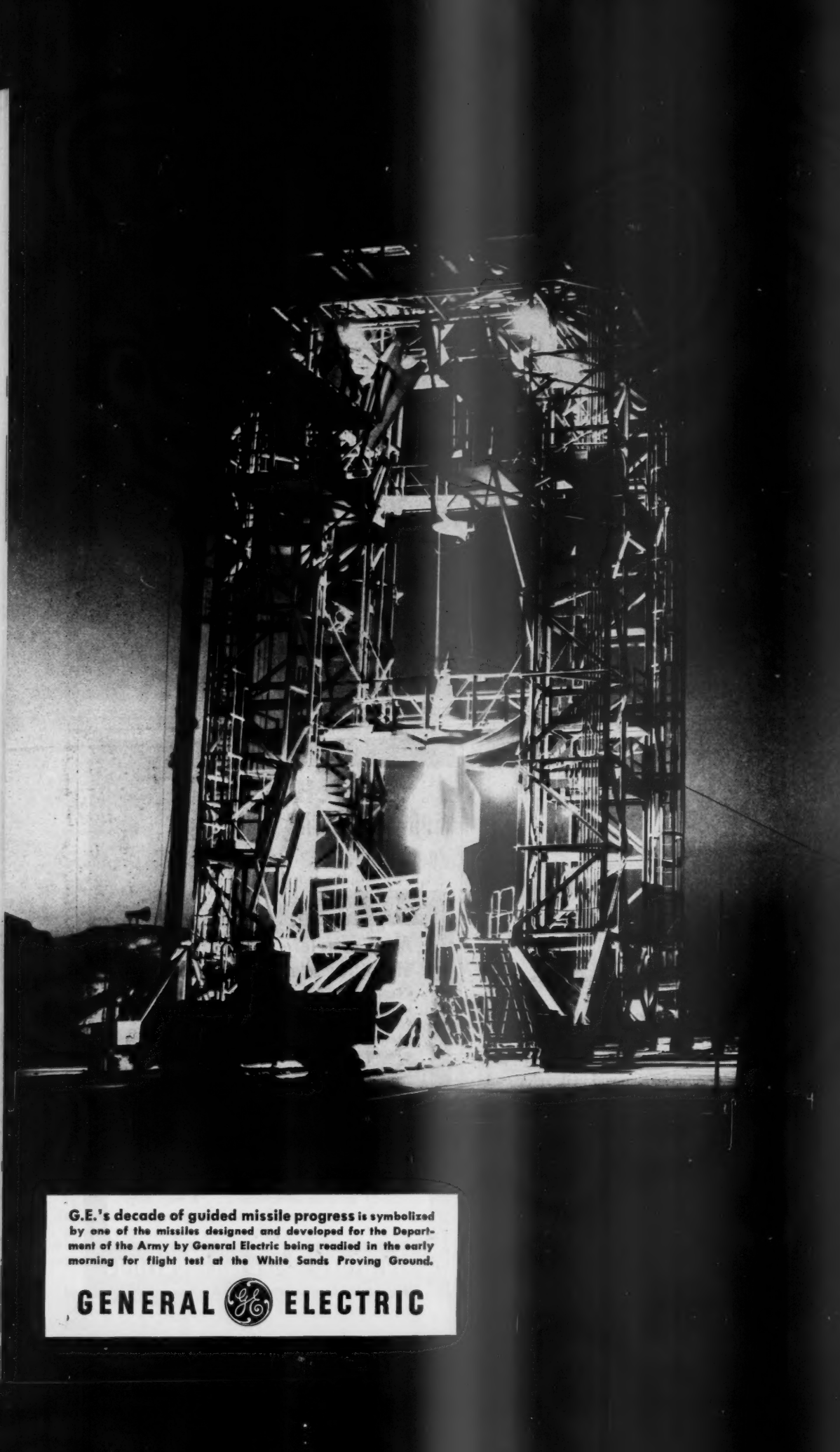
1. **degree in chemical engineering from an accredited school**
2. **minimum of 5 years' industrial experience or equivalent**
3. **some recent experience with solid propellants**
4. **proven leadership and report writing ability**

Expanding division of firm engaged in development and pilot line production of solid propellants for boosters, jatos, rockets, and gas generators.

Initial reply should include qualifications and salary requirements.

Thiokol Chemical Corporation
Elkton Division
Elkton, Maryland
Attn: Mr. D. W. Kershner





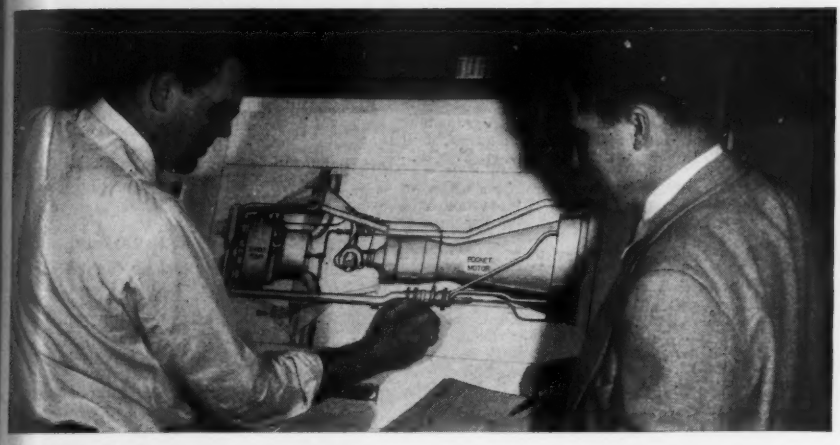
G.E.'s decade of guided missile progress is symbolized by one of the missiles designed and developed for the Department of the Army by General Electric being readied in the early morning for flight test at the White Sands Proving Ground.

GENERAL ELECTRIC

How G.E.'s decade of guided missile progress can aid your missile project

Starting on November 15, 1944, General Electric scientists, following on the heels of our advancing armies, began a study of German rocket development. Since that date, G-E engineers and scientists, working closely with the Army Ordnance Corps, have obtained much guided missile experience.

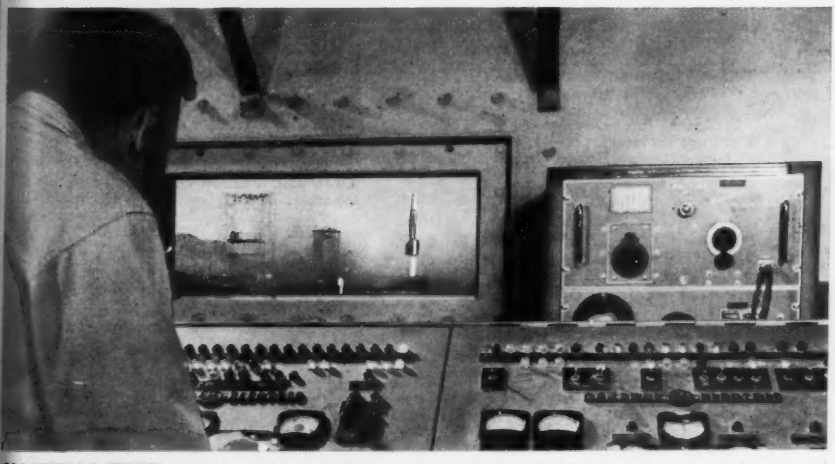
This experience is available to aid your missile project. Having participated in all phases of guided missile development and production, General Electric is well equipped to handle any missile problem. G.E.'s range of experience includes—Propulsion—Guidance—Systems Test—Air Frame.



PROPULSION The highest specific impulse in the history of rocket flight was obtained by a G-E designed rocket engine—a preview of more powerful rocket engines for the guided missiles of the future.



GUIDANCE General Electric's contributions in programmed, radio and inertial guidance have helped industry provide our Armed Services with reliable guidance systems.



SYSTEMS TEST Over one hundred surface-to-surface missiles, designed for guided missile research and development, have been flight tested by General Electric personnel for the Army Ordnance Corps.



AIR FRAME Problems in flying at record speed of 5,000 mph and record altitude of 250 miles were encountered in Project Bumper, where a Wac Corporal rocket was "bumped" from a V-2 in full flight.

For detailed information on how G.E.'s experience in complete guided missile systems as well as sub-system development and production can aid your missile project, contact your nearest G-E Apparatus Sales Office. Section 224-2, General Electric Company, Schenectady 5, N. Y.

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ARS News

33 Papers, Space Flight Symposium, Honors Night Dinner to Highlight Ninth Annual Convention, Nov. 30–Dec. 2

NINE technical sessions, including a Space Flight Symposium to be chairmanned by Milton Rosen, Head of the Rocket Development Branch at the Naval Research Laboratory, will feature the Ninth Annual ARS Convention at the Hotel McAlpin in New York from Tuesday through Thursday, Nov. 30–Dec. 2.

Climaxing the meeting will be the Honors Night Dinner on Thursday evening, Dec. 2, at which the principal speaker will be Major General Leslie E. Simon, Chief of Research and Development for U. S. Army Ordnance.

The complete tentative program is as follows:

TUESDAY, NOVEMBER 30

9:00 A.M. Registration
9:30 A.M. Business Meeting
12:00 Noon Section Luncheon

2:30 P.M. Session 1-A

CHEMICAL COMPONENTS

Chairman: C. F. Warner, Purdue University, Lafayette, Ind.
Vice-Chairman: A. G. Lundquist, O.N.R.

On the Thermal Decomposition of Nitromethane—K. H. Mueller, Aerojet-General Corp., Azusa, Calif. (150-54)
Hydrazine Logistics—D. W. Ryker, Matholin Corp., Baltimore, Md. (152-54)
Newer Techniques in High Pressure H_2O_2 Transfer—N. S. Davis, Jr., R. Bloom, Becco Chemical Division, Buffalo, N. Y. (174-54)

2:30 P.M. Session 1-B

COMBUSTION

Chairman: Lowell Miller, North American Aviation, Inc.
Vice-Chairman: B. Gammon, Lewis Flight Propulsion Lab., NACA, Cleveland, O.

Combustion in Turbulent Streams—W. Berl, J. Rice, P. Rosen, Johns Hopkins University, Baltimore, Md. (155-54)
The Structure and Propagation Characteristics of Turbulent Flames in High Speed Flow—M. Summerfield, S. Reiter, V. Kebely, R. Mascolo, Princeton University, Princeton, N. J. (156-54)
Incipient Flame Propagation in a Turbulent Stream—H. Olsen, E. Gayhart, Johns Hopkins University, Baltimore, Md. (158-54)
The Effects of an Opposing Jet on Flame Stability—Allan Schaffer and Ali Bulent Cambel, Northwestern University, Evanston, Ill. (182-54)

WEDNESDAY, DECEMBER 1

9:30 A.M. Session 11-B

TESTING

Chairman: R. W. Mayer, General Electric Co., Schenectady, N. Y.
Vice-chairman: Walter Kleczek, General Electric Co.

A Versatile Ignition Delay Tester for Self-Igniting Rocket Propellants—M. A. Pino, California Research Corp., Richmond, Calif. (159-54)
Methods for Environmental Testing of Rocket Engines—E. Harslem, L. Muller, Reaction Motors, Inc., Rockaway, N. J. (161-54)
Track Borne Test Vehicles—D. Arenson, Cook Research Lab., Cook Electric Co., Skokie, Ill. (169-54)
Supersonic Component Test Missile—D. Arenson, Cook Research Lab., Cook Electric Company, Skokie, Ill. (170-54)

9:30 A.M. Session 11-C

DESIGN

Chairman: John Sloop, Lewis Flight Propulsion Lab., NACA, Cleveland, O.
Vice-Chairman: A. O. Tischler, Lewis Flight Propulsion Lab., NACA, Cleveland, O.

Some New Aspects of Liquid Bipropellant Rocket Engine Design—H. Jansen, Bell Aircraft Corporation, Buffalo, N. Y. (165-54)
Effects of Ambient Pressure Oscillation on the Disintegration and Dispersion of a Liquid Jet—C. C. Miesse, Aerojet-General Corp., Azusa, Calif.
Rocket Simplification with Multi-Function Components—B. Ellis, Reaction Motors, Inc., Rockaway, N. J. (168-54)
Flow Separation in Supersonic Nozzles—J. W. Drebinger, J. O. Crum, Hughes Aircraft Company, Culver City, Calif. (164-54)

2:30 P.M. Session 111-B

ROCKETS

Chairman: Dwight Gunder, Cornell University, Ithaca, N. Y.
Vice-Chairman: Robertson Youngquist, Reaction Motors, Inc., Rockaway, N. J.
Supersonic Research Sleds and Track Facilities—F. W. Thiele, North American Aviation, Inc., Downey, Calif. (172-54)
Development of a Stabilization System for the Viking Rocket—N. E. Felt, Jr., Glenn L. Martin Co., Baltimore, Md. (171-54)
Performance Analysis of Short-Range High Velocity Ballistic Rockets—W. O. Bergreen, L. M. Miller, North American Aviation, Inc., Downey, Calif. (173-54)
The Pogo Radar Target—G. Moore, Physical Science Lab., State College, N. Mex. (154-54)

2:30 P.M. Session 111-C

DESIGN

Chairman: John Youngquist, Glenn L. Martin Co., Baltimore, Md.
Vice-Chairman: John deNike, Glenn L. Martin Co., Baltimore, Md.

General Enthalpy - Temperature - Entropy Diagram for Gases and Mixtures Thereof and Some Applications in Rocket Thermodynamic Calculations—W. C. Noeggerath, Lockheed Aircraft Corp., Van Nuys, Calif. (175-54)
Determination of Transient Temperatures and Heat Transfer at a Gas-Metal Interface Applied to a 40 mm. Gun Barrel—W. H. Giedt, Detroit Controls, Redwood City, Calif.
A Program for Vibration Control—B. Levine, G. Christopher, General Electric Company, Schenectady, N. Y. (179-54)
Static Seals for Missile Applications—R. R. Ashmead, North American Aviation, Inc., Downey, Calif. (178-54)
Static Seal for Low Temperature Fluids—S. E. Logan, General Electric Co., Inc., Schenectady, N. Y. (176-54)

THURSDAY, DECEMBER 2

9:30 A.M. Session IV-B

COMBUSTION

Chairman: Martin Summerfield, Princeton University, Princeton, N. J.
Vice-Chairman: Ernest Mayer, Arde Associates, Newark, N. J.

Rocket Motor Instability Studies—K. Berman, S. Cheney, Jr., General Electric Company, Schenectady, N. Y. (180-54)
A Monograph on the Problem of Combustion Instability in Liquid-Propellant Rocket Motors—C. C. Ross, P. Datner, Aerojet-General Corp., Azusa, Calif. (181-54)
The Influence of Turbulence on Flame Propagation Rates—R. E. Bolz, H. Burlage, Jr., Case Institute, Cleveland, O. (153-54)

9:30 A.M. Session IV-C

SPACE FLIGHT

Chairman: Morton Gerla, W. L. Maxson Corp., New York, N. Y.
Vice-Chairman: Oscar E. Holt, W. L. Maxson Corp., New York, N. Y

The Incremental-Step Rocket in Free Space—H. R. Wahlin, Bell Aircraft Corp., Buffalo, N. Y. (184-54)
Preliminary Design Study of a Three-Stage Satellite Ferry Rocket Vehicle with Piloted Recoverable Stages—D. Romick, J. Van Pelt, R. Knight, Goodyear Aircraft Corp., Akron, O. (186-54)
High Altitude Launching of a Small Orbital Vehicle—K. Stehling, Bell Aircraft Corp., Buffalo, N. Y. (187-54)

2:30 P.M. Session V

SPACE FLIGHT SYMPOSIUM

Chairman: Milton Rosen, Naval Research Laboratory, Washington, D. C.
Vice-Chairman: Richard B. Snodgrass, Naval Research Laboratory, Washington, D. C.

Instrumentation for Space Vehicles—George W. Hoover, Office of Naval Research, Washington, D. C. (157-54)
Orbits and Lifetimes of Minimum Satellites—S. Fred Singer, Univ. of Maryland, College Park, Md. (160-54)
Importance of Mixture Ratio Control for Large Rocket Vehicles—Rudolph H. Reichel, Bell Aircraft Corp., Buffalo, N. Y.
Geodetic Triangulation From an Unmanned Orbital Vehicle by Means of Satellite Search Technique—Clyde W. Tombaugh, White Sands Proving Ground, N. Mex. (166-54)
Logistic Aspects of Orbital Supply Systems—Wernher von Braun, Redstone Arsenal, Huntsville, Ala. (185-54)

7:00 P.M.

HONORS NIGHT DINNER

Chairman: Andrew G. Haley, President ARS
Speaker: Major Gen. Leslie E. Simon, Chief of Research & Development, U. S. Army Ordnance, Washington, D. C.
Subject: An Appraisal of Postwar Advances in Missiles.

Awards:

Robert H. Goddard Memorial Award to *A. M. O. Smith*, Douglas Aircraft Co., El Segundo, Calif.

C. N. Hickman Award to *H. W. Ritchey*, Thiokol Chemical Corp., Huntsville, Ala.

G. Edward Pendray Award to *Martin Summerfield*, Princeton University, Princeton, N. J.

James H. Wyld Memorial Award to *Milton Rosen*, Naval Research Laboratory, Washington, D. C.

Astronautics Award to *Theodore von Kármán*, Advisory Group for Aeronautical Research & Development, NATO, Paris, France.

Fellow Members:

S. K. Hoffman, North American Aviation, Inc., Downey, Calif.
Lawrence D. Bell, Bell Aircraft Corp., Buffalo, N. Y.
E. N. Hall, USAF, Western Procurement District, Inglewood, Calif.
Clyde Tombaugh, White Sands Proving Ground, N. Mex.
Elmer Wheaton, Douglas Aircraft Corp., Santa Monica, Calif.
A. C. Slade, American Rocket Society, New York.
K. J. Bossart, Convair, San Diego, Calif.
L. D. White, General Electric Co., White Sands Proving Ground, N. Mex.





ARS members and guests, seated a quarter mile away from the 500,000-lb test stand (*right*), watch the 110-sec firing of large rocket

Rocket Firings, Missile Sessions Feature El Paso-White Sands Meeting

VIRTUALLY every rocket and guided missile activity in the United States, and several from foreign countries, were represented among the 300 registrants at the ARS Fall Meeting, Sept. 22-24, at El Paso-White Sands Proving Ground.

Host for the meeting—first Fall gathering of ARS ever staged—was the New Mexico-West Texas Section. Four technical sessions, held at the Hilton Hotel in El Paso, produced stimulating discussion on such subjects as missile reliability and testing, rocket motor design, cosmic rays, space medicine, and rocket test sleds.

On the final day of the meeting four busloads of ARS members and guests were taken on a tour of White Sands Proving Ground. Feature of the tour was the witnessing of the firing of a large lox-alcohol rocket engine for undisclosed thrust on the 500,000-lb static test stand. Duration of firing (above) was 110 sec.

At the close of the technical sessions on Thursday evening, Sept. 23, a banquet was held. The principal speaker, Brig. Gen. W. L. Bell, Commanding General of WSPG, gave listeners a preview of the next day's tour of the famed missile test center and showed a film, declassified for the occasion, "An Introduction to White Sands Proving Ground."

Toastmaster of the banquet, ARS President Andrew G. Haley, expressed gratitude to the New Mexico-West Texas Section for the fine effort put into the organizaton of the meeting. Particular appreciation was bestowed upon New Mexico-West Texas president Frank L. Koen; board chairman Edward Francisco, supervisor of the WSPG tour and founder of the NM-WT Section; H. L. Karsch; meeting chairman A. T. Finney; and secretary-treasurer G. Harry Stine.

In addition to the large rocket firing, the tour of WSPG included the witnessing of a 25-sec firing on the 100,000-lb test stand of red fuming nitric acid-aniline engine; inspection of camera and radar stations; inspection of the Aerobee, Viking, Corporal, Honest John, and Nike missiles and launching facilities; and explanation of the missile flight safety program which insures that test "birds" will stay within the range of the proving ground. A barbecue luncheon was then served at the officers club.

Following is a brief summary of the papers presented at the sessions:

Generation of High Gas Pressure Through Hydraulics (136-54). H. H. Halperin of Greer Hydraulics, Inc., Jamaica, N. Y., gave details of a portable pressure booster for maintaining a minimum of 6000 psi pressure on missile systems during fueling.

Test Procedures vs. Test Instruments (137-54). Robert Lucas of Bell Aircraft Corp., Holloman Air Force Base, N. Mex., outlined a philosophy which he felt should be applied to the missile industry. It accents the need for distinguishing between where test procedures should apply and where test equipment should apply in a missile system.

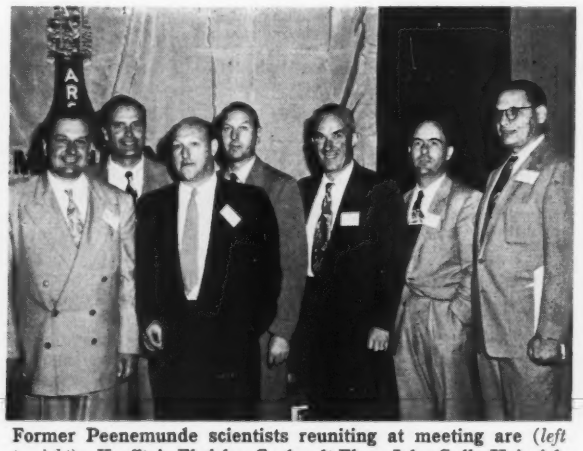
Use of Rockets for Cosmic Ray Research (138-54). R. E. McDaniel of WSPG described the study of cosmic rays above the earth's atmosphere by the exposure of nuclear emulsions in rockets flown at WSPG.

Review of Biological Effects of Subgravity and Zero-Gravity (139-54). D. G. Simons of Holloman Air Development Center told of experiments with animals and humans which give evidence of disorientation and disco-ordination due to exposure to subgravity and zero-gravity conditions. He cited the important function which the vestibular apparatus has in controlling physiological and psychological responses to such exposure.

Rocket Sleds as Biological Test Vehicles (141-54). John P. Stapp of Holloman reviewed a series of 215 experiments on sleds propelled by solid propellant rockets along a railroad track. Humans, chimpanzees and anthropomorphic dummies are the subjects studied during subjection to windblast, tumbling and prolonged deceleration.



At the El Paso-White Sands meeting. Posed under Nike missile in Hilton lobby are (*left to right*): E. E. Francisco, NM-WT Board chairman; A. G. Haley, ARS president; F. L. Koen, NM-WT president; Ed Brown, NM-WT vice-president; G. H. Stine, NM-WT secretary-treasurer; J. J. Harford, ARS executive secretary



Former Peenemunde scientists reuniting at meeting are (*left to right*): Krafft A. Ehricke, Gerhardt Eber, John Gelb, Heinrich Schultze, Kurt Steinhoff, Helmut Horn, and Konrad Dannenberg. Ehricke is at Bell Aircraft; Gelb is a Cleveland consultant; Dannenberg is at Redstone Arsenal; Horn at WSPG; others are from Holloman



Three of the authors converse after session at El Paso-White Sands meeting. They are Col. John P. Stapp, Robert Lucas, and Major D. G. Simons



Presenting award to ARS President Haley, on behalf of IAF, is Franco Fiorio, assistant air attaché of the Italian Embassy. Brig. Gen. W. L. Bell, main speaker at the dinner, looks on.

Free Flight Determination of Missile Environment Generated by Solid Propellant Motors (140-54). H. Gumbel of the U. S. Naval Air Missile Test Center, Point Mugu, Calif., analyzed the advantages of free flight testing and pointed out methods for determining the vibratory forces created by the burning of solid propellant motors. He also discussed the design of an instrumentation package which can transmit test information at frequencies up to 20,000 cps.

Radar and Guided Missiles (143-54). W. Hiltz of Holloman provided a general résumé of the principles and techniques used in applying radar to guided missile systems for providing positioning information and for linking the missile to a ground control system.

Ascent of Orbital Vehicles, and **On the Mechanics of Descent to a Celestial Body** (146-54). In the first of these two papers by Krafft A. Ehricke of Bell Aircraft Corp., of Buffalo, N. Y., the ascent into a satellite orbit along a minimum energy ellipse was compared to ascent along an elliptic trajectory. Conclusion is that more powerful stages would be required if a ballistic ascent is used.

The second paper dealt with descent which does not follow great circle flight; pointed out the difficulties encountered; then showed how compensating steps can be taken.

General Considerations in Missile Evaluation (147-54). R. Weller of USNAMTC, Point Mugu, outlined the problems encountered at guided missile test ranges in connection with missile evaluation, such as: investigation of missile reliability, requirements for adequate ground testing, planning of experimental programs, the significance of data, and the necessity for detailed measurement of flight conditions and their simulation in the laboratory.

System Simulation for Missile Evaluation (148-54). Using a hypothetical missile, John H. McLeod, Jr., of USNAMTC, Point Mugu, showed how laboratory simulation before flight test can determine "sensitive" parameters in which small variations in tolerances can significantly change system performances. Laboratory simulation is also employed *after* flight test to interpolate or extrapolate system performance to conditions other than those of the test involved, or to reconstruct the operation so as to reproduce symptoms and explain the behavior of expended components.

Reliability in Guided Missiles (149-54). Citing experiences with the V-2, the Viking, and the Bumper rockets, R. P. Haviland of General Electric, Key West, Fla., launched a discussion of general steps and procedures required to produce optimum reliability in components and missiles.

Rocket Motor Design Principles (144-54). Robert Gordon of Aerojet-General Corp., Azusa, Calif., conceived a motor of infinite thrust as a means of establishing design parameters for large thrust units.

Reliability Concepts and Methods in the Evaluation and Improvement of Guided Missile Systems (preprint not available.) J. R. Duffett of WSPG contrasted reliability problems of piloted aircraft to those of guided missiles.

A Feasible Method of Tracking Orbital Vehicles (preprint not available). Clyde W. Tombaugh of WSPG described his "tracking Schmidt" optical device which can photograph a V-2 at a distance equal to that of the moon from the earth. Used in Tombaugh's satellite search program, the unit has an 8½-in. aperture and a 13-in. focal length.

How to Beat the Engineer Shortage

THE rapidly expanding rocket and jet propulsion field is probably suffering more than any other from an insufficiency of qualified scientists and engineers. That's why the efficient utilization of available talent is so important, and why the New York Section chose to devote its June 25 meeting to a panel discussion: "The Administration of Technical Projects."

On the panel, moderated by *Harry B. Horne,* executive vice-president of Wallace Aviation Corp., Wallingford, Conn., were *D. B. Clark,* Manager of Advanced Design at Wright Aeronautical Div., Curtiss-Wright Corp., Wood Ridge, N. J.; *E. M. Fettes,* Chief of Research and Development at Thiokol Chemical Corp., Trenton, N. J.; *R. L. Graupe,* Chief Project Engineer, Development, at Fairchild Engine & Airplane Corp., Engine Div., Farmingdale, N. Y.; and *J. P. Layton,* Chief Jet Propulsion Engineer, James Forrestal Research Center, Princeton University, Princeton, N. J.

In spite of the diverse backgrounds of the panelists, basic agreement was obvious on methods of getting good administration.

Major points brought out were:

* Project engineers should participate strongly in getting contracts—in defining the jobs, estimating costs and man-hours and even in direct selling in some cases—in order to develop their management sense.
* Good administration is possible in all forms of organization of projects—line, line-and-staff, specialist grouping, and project team.
* The line system offers quick results if manpower and money are plentiful, but should not continue beyond the initial phases of the project, otherwise it causes "empire-building."
* Line-and-staff organization is cheaper in dollars and man-hours but takes longer in elapsed time to complete the project.
* Good budget control helps prevent the problem of excessive test pieces, free samples or other extra work for sales purposes by showing management the cost of such work. It also eliminates excessive side-road explorations.

Some of the interesting remarks made during the discussion were:

Graupe: "Some managements, because of poor planning, dump a project in the lap of the Chief Engineer, and he has to figure out who he can swipe from another project to do the job."

Layton: "Most organizations need more meticulous planning—even of fundamental research. If professors would only understand that we're not after their academic freedom, simply a schedule."

Clark: "Projects should be set up as realistic targets—a balance between the conservatism of the project engineer and the optimism of the salesman."

The panel was organized by C. W. Chillson, Program Chairman of the Section.

1955, ARS 25th Year, May See Seven Meetings

NEXT year will mark the 25th anniversary of the founding of the AMERICAN ROCKET SOCIETY and will be featured by a program which may include participation in seven national and international meetings. Scheduled are the following:

Jan. 24–28	New York	All-day session on JATO at IAS Annual Meeting
Apr. 18–22	Baltimore	Three sessions at ASME - ARS Spring Meeting
Nov. 13–18	Chicago	Tenth Annual Convention

It is also probable that the Society will participate in the ASME Semi-Annual Meeting in Boston, June 19–23; in a Combustion Meeting at Northwestern University, Evanston, Ill., in late August; and that a three-day Fall Meeting, similar to this year's El Paso-White Sands Proving Ground gathering, will be held sometime in September.

The Sixth IAF Congress will be held in Copenhagen, probably in August.

Papers, or abstracts of proposed papers, for any of these meetings (except the January one, the program for which is complete) should be sent to the National Office at least four months prior to the meeting date.

Section Doings

Central Texas. Officers of this newly formed Section, which received its charter from President Haley in September, are Barnet R. Adelman, president, and R. C. Terry, secretary, both of Phillips Petroleum Co., McGregor.

Cleveland-Akron. Fernand Florio, chief engineer of a French rocket company, SEPR, and Georges Delval of SNCASO, French aircraft manufacturers, spoke of the new rocket-powered interceptor, the Trident, made by the two companies, at the September meeting. Present version is powered by a 10,000-lb thrust engine mounted in the fuselage. Fuel is a mixture of xylidine, methanol, and furfuryl alcohol while the oxidant is white fuming nitric acid. The airplane made its first flight recently. It carries two turbojet engines in the wings for sustained flight.

New York. In spite of bad weather, almost 300 members and guests visited the U. S. Naval Air Rocket Test Station at Lake Denmark, N. J., on September 18. An exhibit showing test equipment, electronic equipment, protective clothing, etc., was set up, and there were several demonstrations of ignition experiments in addition to an exhibit of various motors including a German V-2 engine. After a welcome by Captain W. G. Jackson, Jr., USN, Commanding Officer, a newly declassified sound film on tactical transport, service, and launching of the V-2 missile was shown, introduced by Bertram N. Abramson, Head Engineer. This was followed by a description of Reaction Motors, Inc., test facilities by Eric Harslem, Chief Test Engineer at RMI, who also introduced a color film on the Viking missile. After a box lunch, the meeting moved to the test facilities to see a 1000-lb solid propellant rocket fired. The meeting concluded with a demonstration of fire fighting techniques by the fire department of NARTS.

Northeastern New York. Fernand Florio and Georges Delval discussed the French SNCASO interceptor at a September 14 informal dinner (see Cleveland-Akron news).

On September 21, Milton Rosen, head of NRL's Rocket Development Branch, told of the problems which have "bedeviled" Viking personnel in testing and launching at WSPG. Engineers at GE's Malta Test Station state that the problems matched those encountered in their own experiences remarkably.

Pacific Northwest. Provisional officers and directors elected to serve this Section are: President, R. Max Clayton; vice-president, William G. Ramroth; secretary, W. Emmett Coon; treasurer, George W. Hettrick; Directors: David H. Suddeth, Nathan S. Ruder, Harold G. Lustig, Robert M. Bridgforth, Jr., Henry L. Turner, Jr., Stanley W. Leszynski, all of Boeing Airplane Company, and James Fisher of the University of Washington.



At the New York Section meeting (*left to right*): **Paul Terlizzi, NARTS engineer; Cmdr. Page Goldbeck, Enf. Officer, NARTS; C. W. Chillson of Curtiss Wright; Michael J. Samek, New York Section president; and Capt. W. G. Jackson, Commanding Officer at NARTS**

Southern California Section. The trip to the U. S. Naval Air Missile Test Center at Point Mugu drew 81 ARS members. Captain E. M. Condra, Commanding Officer and Dr. Royal Weller, Chief Scientist, outlined the Center's activities. Because of low ceiling, flight operations were cancelled, unfortunately. Individual clearances were handled with a minimum of confusion by local security officers.

On September 8, William Bollay and Wayne Ehler of the Aerophysics Development Co., spoke at a roast-beef dinner meeting on "New Horizons Using Rockets."

Major Arthur "Kit" Murray, test pilot for the Bell X-1 and X-5 among other aircraft, spoke at the October 13 dinner meeting at the IAS Building in Los Angeles.

Southern Ohio. Commander R. C. Truax, ARS director, presented a charter to this Section on September 16. Elected to office are: Warren White, president; W. J. Mizen, vice-president; Don E. Wrede, secretary; Robert W. Everson, treasurer. The directors are Richard Lott, C. W. Hosier, J. T. Marshall, H. Rosenberg, and H. R. J. Grosch. Mizen and Marshall are with Bendix Aviation, Hamilton Division; Lott with Reaction Motors, Inc.; Hosier with Aerojet-General Corp.; and Rosenberg with Planet Products. The rest are from General Electric's Aviation Gas Turbine Division, Cincinnati.

Twin Cities. Fourth newly chartered Section is this one with members from the Minneapolis-St. Paul area. Officers are Joseph J. Schons, president; Richard Panek, secretary; David Zipoy, treasurer. Directors are Thomas F. Irvine, Jr., Albert P. Earle, and Jerry Koch. All are at the University of Minnesota as students or instructors.

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The Rocket Test Engineer

by Bruno Brennschluss[1]

ALTHOUGH the rocket motor of today has been perfected to such a degree that 99 per cent of them do not need testing, it was nevertheless decided to test 1 per cent of the rocket motors so that information from the tests could apply to the other 99 per cent in saying that 99 per cent of them didn't need testing. Amidst this environment the rocket test engineer emerged.

Description of the Genus

Theoretically only one type exists; however, due to the selective process of dynamic evolution, four types are known to exist at various rocket establishments.

The IDEAL type is the rocket test engineer you see in the movies, ads, and press releases. He is usually clad in protective clothing, wears a face shield, and sometimes sports a supersonic dunce cap. Invariably he is described as a refugee from Mars. This type has an extremely short half life, and immediately after being photographed he disintegrates into one of the types described below. It is well known that he is well grounded in rocketry.

The CALIFORNIA type is in the 20–25 year bracket. He wears an open-necked sport shirt at least 135 decibels louder than a typical rocket shot. His baseball cap is always perched on his head at an angle of at least 45 degrees or more, and by his side is always found a copy of "Jet Comics" and a frosty mint julep. He is quite versatile in the use of artifices.

The MISFIRE type being in the 25–35 year niche obviously has been around rockets a little more than his junior Californian type. He invariably slinks around in chartreuse sneakers and plaid shirts 13.5 decibels louder than a rocket firing. His tie is a motley color, stained by eggs, soup, and red fuming nitric acid. Wandering about the installation (gutturally cursing the project engineer with quaint four-letter words) he is usually carrying a gnarled hydraulic line on some mission no one has been able to discover. That bulge in his hip pocket, he claims, is only used to prime the alcohol pump before firing.

The SENIOR type takes superadequate measures for tests. When not in his foxhole he can be detected by that light streaming through the jagged hole in his head. Besides knowing all about rockets, he is an expert on all kinds of soil and on certain internal propellants.

Modus Operandi

To most persons (especially to the brass) the rocket test engineer has it easy—all he does is press the firing button. This is true. The only time he appears to work is when the brass is around. He calls up the big wheel in the morning and announces a firing at 3:30, and then conducts elaborate pressure and thrust calibrations, checkouts, and recalibrations. Fatigued by all this precise scheduling, he takes two hours for lunch, and an extra hour talking to that cute hat-check girl at the local cocktail lounge. With only fifteen minutes before the big firing he calls up the big wheel to postpone the shot to 4:30 to impress him with the enormity of the task. The brass finally arrive, after a delightful talk with that cute hat-check girl at the local cocktail lounge.

Does he fire? No! With only the hum of electronic equipment he performs the sacred ritual of a buildup. Down-range stations report in; warning sirens, horns, and buzzers blare; lights flash, warning rockets are fired, and finally everything becomes quiet; the test engineer's announcement of prefire time comes in through the audio system. Then comes the elaborate count-down. By this time the brass have become so nervous that they have chewed off all their finger nails. Then, in order to prevent serious damage to the brass, the test engineer finally fires the rocket. After a blast of up to 60 sec all becomes quiet again.

The score is not impressive: only 60 sec out of 28,800 sec per day are devoted to firing. Don't be mistaken, though. The fame of a rocket test engineer is inversely proportional to the number of motors fired. Thus, a small-time operator may boast of firing 200 motors per day, while the real expert will spend at least six months getting set up to fire a unit with a burning time of just 125 millisec.

During the firing the brass observe how efficiently the test engineer functions. He knows each switch and can tell you that the gleaming pilot light next to the second bank of rheostats indicates that coffee is ready. He is continually adjusting meters and can trip hundreds of SPST switches per min. In the evening, after all this, he is lucky to be able to start his car the first time, instead of putting the key in the lighter socket.

Before each test the rocket test engineer knows just what the specific impulse will be. He just goes through the motions merely to please the project engineer. When starting in on a new project he causes all sorts of things to go wrong. The motor won't ignite, it blows up, or, if it does go, the C* is at least 2000 ft/sec below what it should be. Naturally, the project engineer begins to sweat. So, the test engineer makes some nonsensical change, such as sanding that superfinish on the nozzle, or putting a few dents in the injector ports. Finally, the rocket motor purrs sweetly every time the test engineer appears, just to avoid being kicked in the critical section.

Just humor him, and give him a screwdriver and a pair of pliers, and he can accomplish miracles with your motor. If you arouse his wrath, all hell will break loose. It is a well-known fact that a good test engineer can bring a project to an ignominious halt by looking cross-eyed at your rocket motor.

What does a rocket test engineer do when he isn't firing rocket motors? Mostly nothing. Since he is usually engaged in destructive testing, it is alleged that the test engineer is not a creative person. This is nonsense. When he is not destructing, he is constructing. What? Wildly imaginative technical reports and new, diabolical, testing devices.

Basically being quite ignorant, what he lacks in intelligence he makes up for in stupidity. Thus, the test engineer usually describes his work in technical reports. These reports are masses of tables, tabulations, graphs, thrust-time traces, equations, photographs, sketches, histograms, and nomographs bound together with a minimum of words. With a vocabulary of a child of ten, he will clannishly babble about a "hot and sweet" shot, chuffing, hangfire, etc., in such a manner that only another test engineer knows what is being said. When asked why he fired a particular motor, he will honestly say, "Beats the hell out of me," or more exactly, "To see if the goddam thing will blow up"; while the project engineer is likely to place the reason as: "To determine if the new area ratio is sufficient to dampen transient shock oscillations which may be of sufficient amplitude to rupture the wall." The test engineer will write his voluminous reports amid the din of rocket blasts interrupted only by such thoughts as how efficiently he could toast bread in the exhaust flame, dancing molecules of ethanol, or that cute hat-check girl down at the cocktail lounge. His desk, cluttered with nozzles converted into ash trays, vibrates in phase with his pen as he grinds out the reports.

Although the test engineer is always working on a "hot" new gadget to do something or other, most of his "black boxes" are purchased. However, by the time the test engineer gets through with it, the manufacturer will never recognize his product. The oscilloscope is modified to deliver a rounded square wave, relays are put on existing relays, generators (run by batteries) take the place of batteries, and a VHF signal generator is used to toast sandwiches. Occasionally, some of this is necessary.

In the gadget department, the test engineer shines. His control panel with its maze of switches, pilot lights, relays, buttons, meters, jacks, resistors, condensers, coils, chokes, filters, decade boxes, oscillators, speakers (tweeters and woofers), transformers, vacuum tubes, transistors, and miles of wire and tons of solder, is a joy for any electrical supply company. The control panel, like its creator the test engineer, is almost human. Early in the morning it is balky; at noon it needs some hot shots; and an hour before quitting time it needs to be kicked in a few strategic spots to keep operating.

Thus, we have seen how the rocket test engineer is the very cornerstone of our coming age of rocket power. If it doesn't come, you can be sure that it was not the fault of the rocket test engineer!

[1] Correspondence with Mr. Brennschluss should be directed to A. J. Zaehringer, American Rocket Company, Wyandotte, Mich., who is the only one who knows how to reach him.

Book Reviews

C. F. Warner, Purdue University, *Associate Editor*

The Next Fifty Years of Flight, by B. Balchen and E. Bergaust, Harper & Bros., New York, N. Y., 1954, 214 pp. $3.

Reviewed by R. L. Duncan
Patuxent Naval Air Test Center

The role of a prophet is not an easy one and is especially difficult in the field of aeronautics where progress is amazingly rapid. During lunch, dinner, and other times when aviation people get together, the topics of discussion tend toward predictions of the distant future, called "taking a trip to the moon" or "hangar flying." This general practice of dreaming of the future is an interesting diversion for the people involved in research and development who are continually faced with the problem of determining the direction in which our program must go tomorrow or in the immediate future. I believe, therefore, that the authors derived a great deal of pleasure from assemblying the material necessary for their interesting and enlightening book. It is evident that a number of discussions were held between well-known people in aviation and the authors. Men in nearly every division of the aviation field were contacted, including Arthur Godfrey, and their statements and predictions are quoted so that a good cross section of opinion, in addition to that of the authors, is contained in the book, and thus an interesting summary of the present thinking in the industry is given.

Looking back over the progress of the past fifty years and picturing a man in the early years of the century with the courage to predict the accomplishments of the first fifty years of aviation makes me think that such a man would have fallen far short in his prediction of the tremendous developments in aviation. And so it may be that the authors also have missed many a future wonder that cannot be foreseen at present. There is no doubt that many controversial issues have been covered in the book and perhaps, among the experts, a difference of opinion could be found for every prediction. The statements of the authors are to me generally conservative, though many a difficult problem must be solved before the predictions can be realized. The chapter on space travel is exceptionally well written and the discussion of private and commercial flying is quite interesting. The Arctic Story, of course, is an outstanding chapter and could only be so well presented by these authors.

Worlds in Space, by Martin Caidin, Henry Holt and Co., New York, N. Y., 1954, 212 pp. $5.

Reviewed by W. R. Warren, Jr.
Princeton University

Martin Caidin has written a rather comprehensive summary of many current ideas concerning space flight. One of the features of this book is the author's realization of the problems inherent in the majority of proposals that have received wide publicity. He treats these problems with more understanding and less unbounded enthusiasm than is usually the case. The reader, who might at first feel that he is searching through another fantasy, is led to the realization that space travel will actually exist in the future, although, perhaps, to less dramatic ends than he had expected.

The first chapter presents a concise summary of rocket achievements in the past decade and indicates the progress, as well as the lack of progress in certain areas, that has been made toward the various space flight goals. The experience of German engineers and the use of captured V-2 missiles as test vehicles are shown to have been quite important in postwar rocket research.

The following sections deal with immediate aims of space enthusiasts. Manned and unmanned satellites are discussed, their important scientific potentialities pointed out and their overestimated military usefulness. A long chapter is devoted to the difficult problems (weightlessness, cosmic radiation, extreme acceleration) man must face when he attempts to travel beyond the earth's atmosphere. Caidin then gives a cursory examination of existing plans for satellite and space ship construction. In this regard, he takes a more pessimistic, and realistic, viewpoint than other writers concerning both the financial and time estimates.

Lastly, the author treats proposed expeditions to other worlds. Travel to and from the moon receives the most detailed treatment, but some discussion of the practicality and magnitude of effort needed for interplanetary travel in our own solar system is also included.

The more imaginative portions of the book are adequately illustrated by Fred I. Wolff. The historical section is supported by many well-known photographs.

V-2, By Walter Dornberger, The Viking Press, New York, 1954, 281 pages. $15.

Reviewed by C. F. Warner
Purdue University

Much has been written of the technical aspects of the German rocket weapons and their development; but in *V-2*, Dr. Walter Dornberger presents a graphic account of the men who brought the German rockets into being. This book will appeal not only to those directly connected with the field, but also to the nontechnical reader interested in the history of the development of mankind's great inventions.

Walter Dornberger, Doctor of Engineering, General in the German Army, and Head of the Peenemünde Rocket Station captures the reader's interest immediately with a vivid description of the tense moments just preceding and during the firing of the world's first large liquid propellant rocket, the German A-4, later called by the English, the V-2. After the story of the initial A-4 flight has been told, the author returns to the first feeble efforts of the Society for Space Travel (V.F.R.) and from there tells of the ever-expanding German rocket development program halted only by the collapse of the German war effort.

In addition to describing the manifold technical problems successfully overcome during the building of the V-2, the author tells of the many obstacles that beset the Peenemünde technical staff because of the personal jealousies, political rivalries, and shortsightedness of Hitler, Himmler, Goering, Degenkolb, and others.

At the close of the book the reader is left with the feeling that, most fortunately for the Allies, the complete German rocket program can be characterized by the famous words "too little, too late."

Airplane Structures, Volume I, 4th edition, by A. S. Niles and J. S. Newell, John Wiley & Sons, Inc., New York, N. Y., 1954, 607 pp. $7.75.

Reviewed by J. P. Layton
Princeton University

Professor Niles and the late Professor Newell have thoroughly revised volume I of their well-known two-volume work, "Airplane Structures," although it continues to be devoted to the elementary principles underlying all structural design. The fourth edition of volume I should continue to be widely used as a text in teaching aeronautical structures, but its usefulness should not be neglected in the design of efficient structures in other fields, as the desirable trend toward high strength weight ratios becomes increasingly recognized.

Text and problems have been redone and the following new material has been added: (1) a new approach to the analysis of riveted joints; (2) a chapter on the physical properties of structural materials; (3) a treatment of combined bending and torsion of a shell with constant-shear-flow webs; (4) an analysis of a space truss with members subjected to bending moments in addition to axial loads; (5) new monograms for use in practical design work; (6) the NACA method for the analysis of incomplete tension-field beams; and (7) a new and simpler derivation of the Euler formula. Some of the material from volume II has been incorporated into this edition of volume I, primarily a chapter on the Maxwell-Mohr method for the analysis of statically indeterminate structures.

Since a single work cannot provide all things, some of the statements and material do not correspond to current structural practices or reflect light into the darker corners of present need. However, the basic organization and clear logic of the book remain as a sure fundamental base from which to launch onto the stormy sea of structural analysis.



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Wind Tunnel Testing, second edition, by Alan Pope, John Wiley & Sons, Inc., New York, N. Y., 1954, 511 pp. $8.50.

Reviewed by SEYMOUR M. BOGDONOFF
Princeton University

Mr. Pope has done an excellent job of revising, correcting, and bringing up to date his first edition of "Wind Tunnel Testing." The book is a very good introduction to subsonic wind tunnel testing and design and, as such, completely fulfills the stated purpose of the author. In this field, the text is authoritative, quite complete, and well illustrated. The numerous problems are especially useful for undergraduate instruction. The sections on testing procedures and wall corrections have been considerably improved in this edition.

The author has added sections on transonic, supersonic, and hypersonic testing to his first edition. These sections have added considerably to the value of the book, but they are not anywhere near as complete or authoritative as the low-speed sections. In the light of the fact that many schools introduce such subjects in undergraduate instruction and many engineers in the aeronautical industry are working on transonic and supersonic applications, the reviewer feels that a more complete treatment of the problems in these ranges is needed. The inclusion of this material may prove to be difficult because of the introductory nature of the presentation and the problems of classification of some of this new material.

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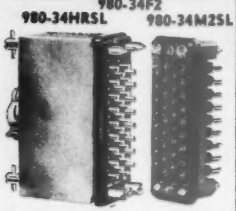
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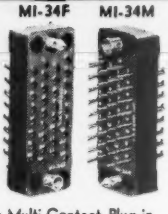


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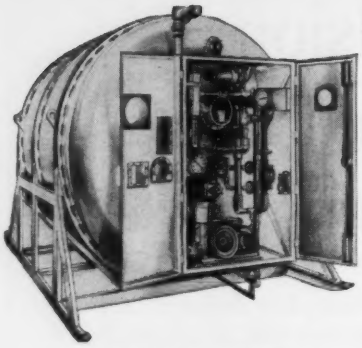
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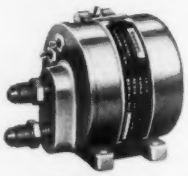
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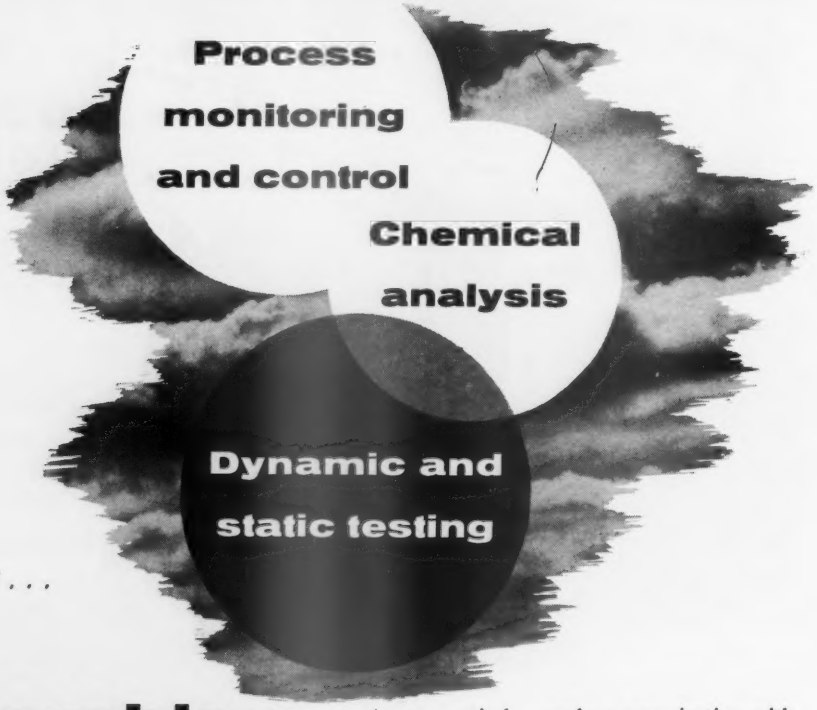
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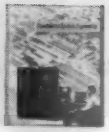
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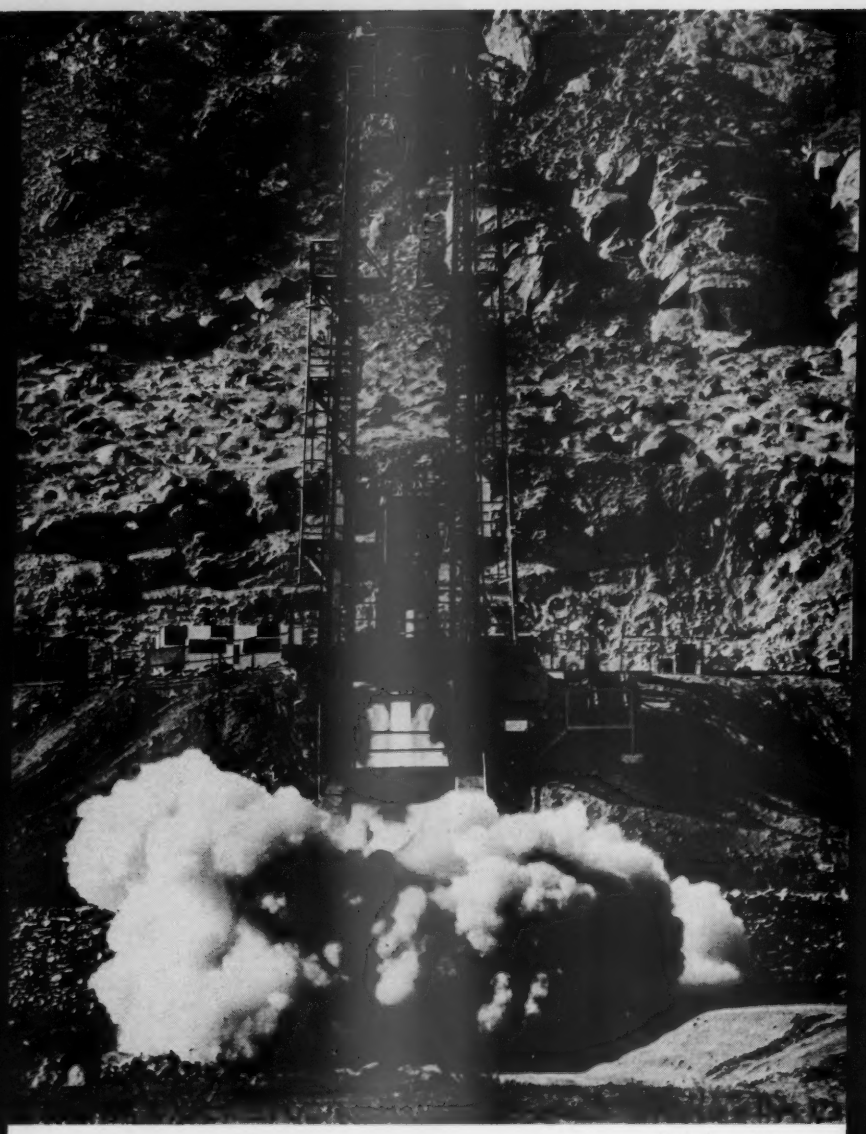
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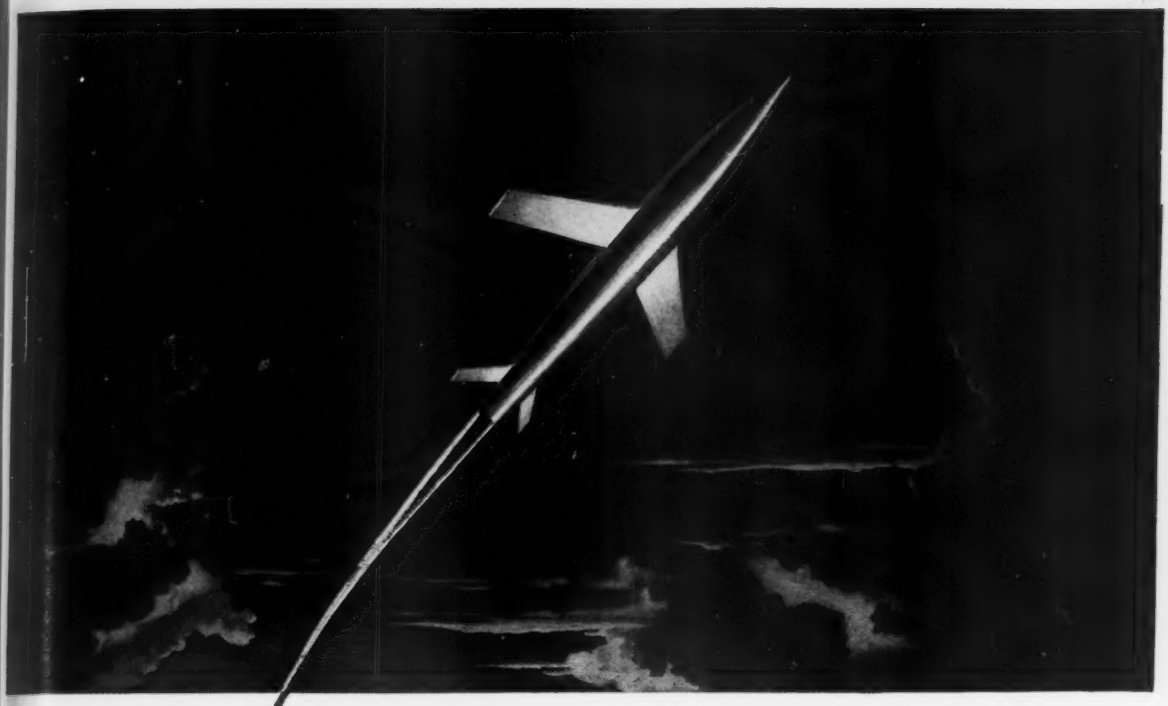


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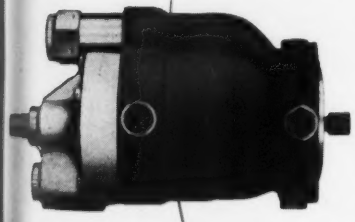
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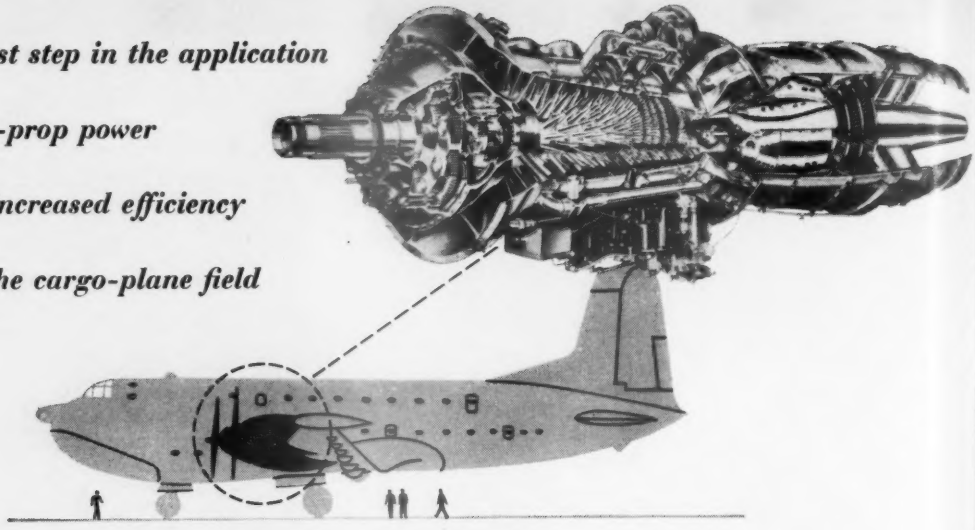
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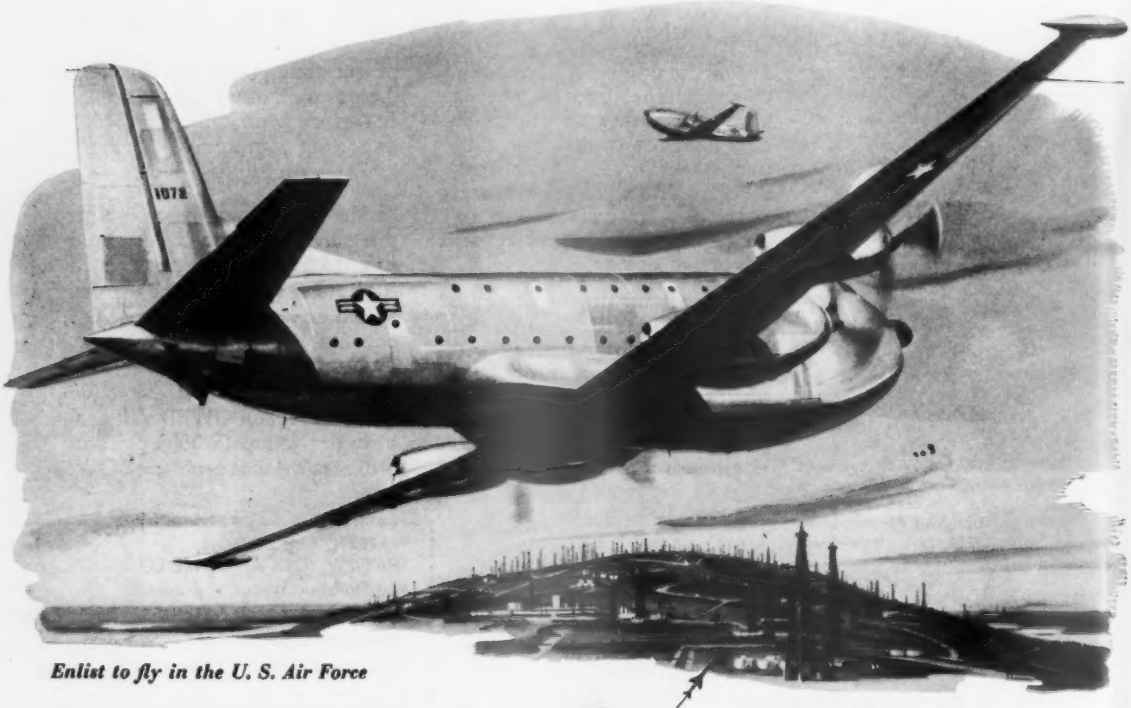


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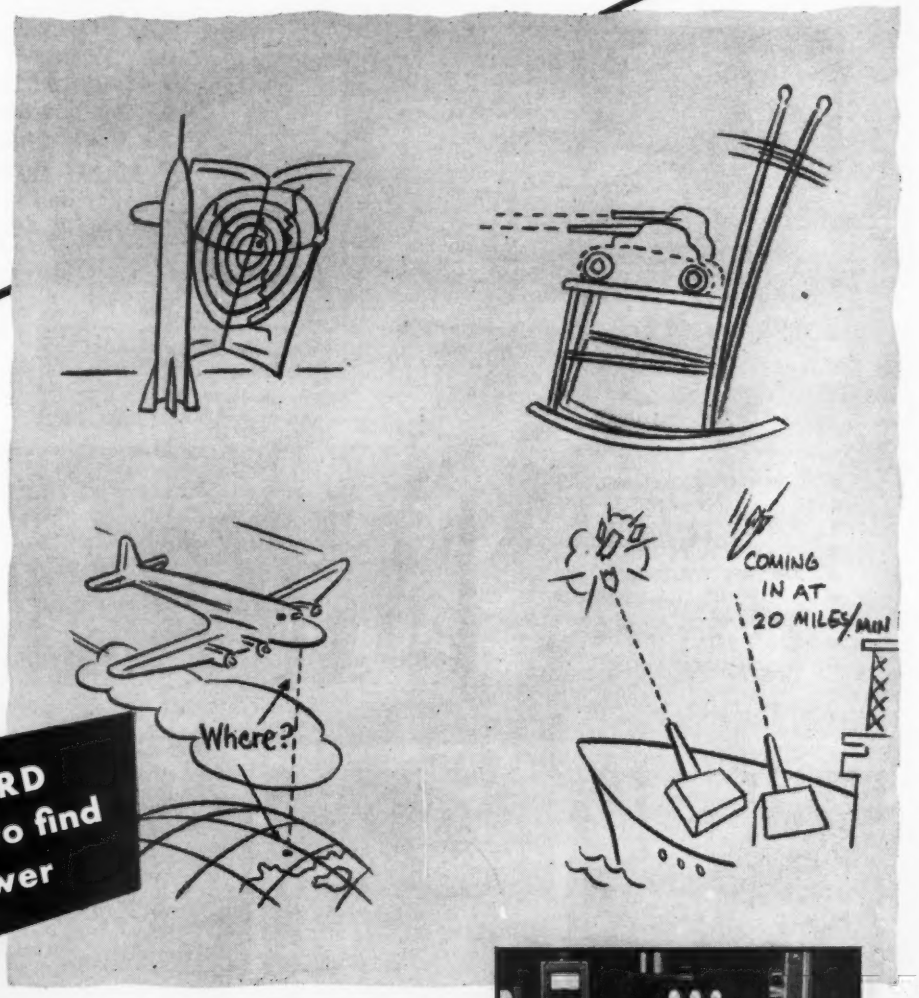
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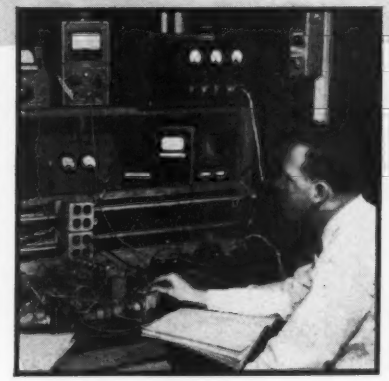
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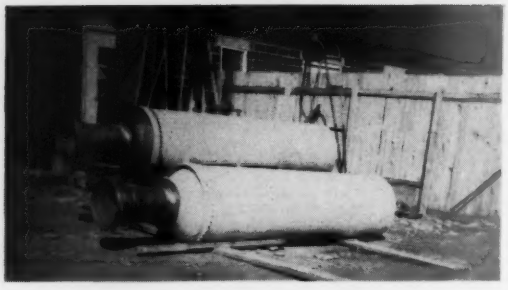
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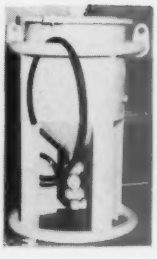
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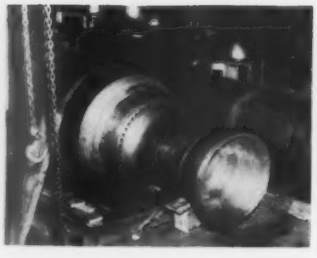


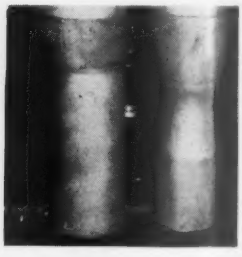
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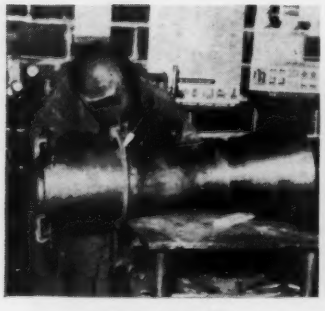
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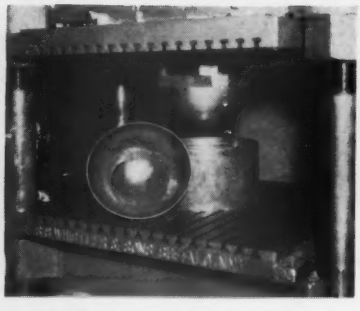






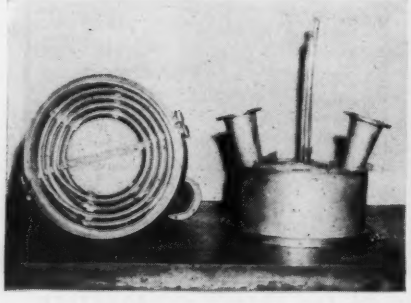
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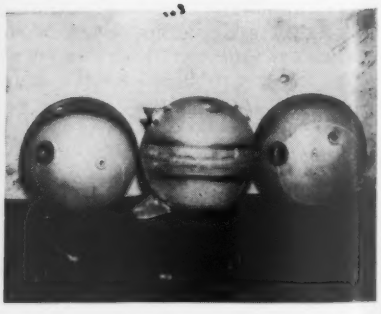




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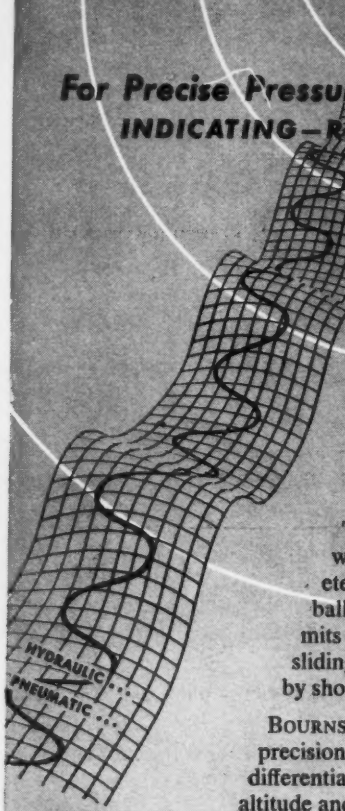
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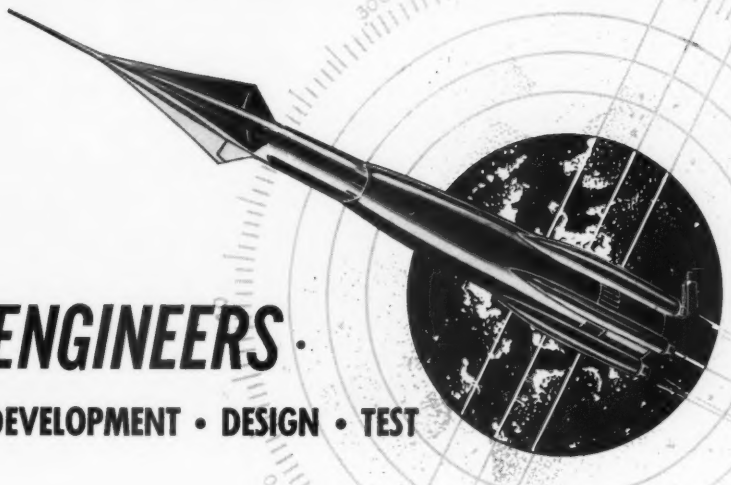
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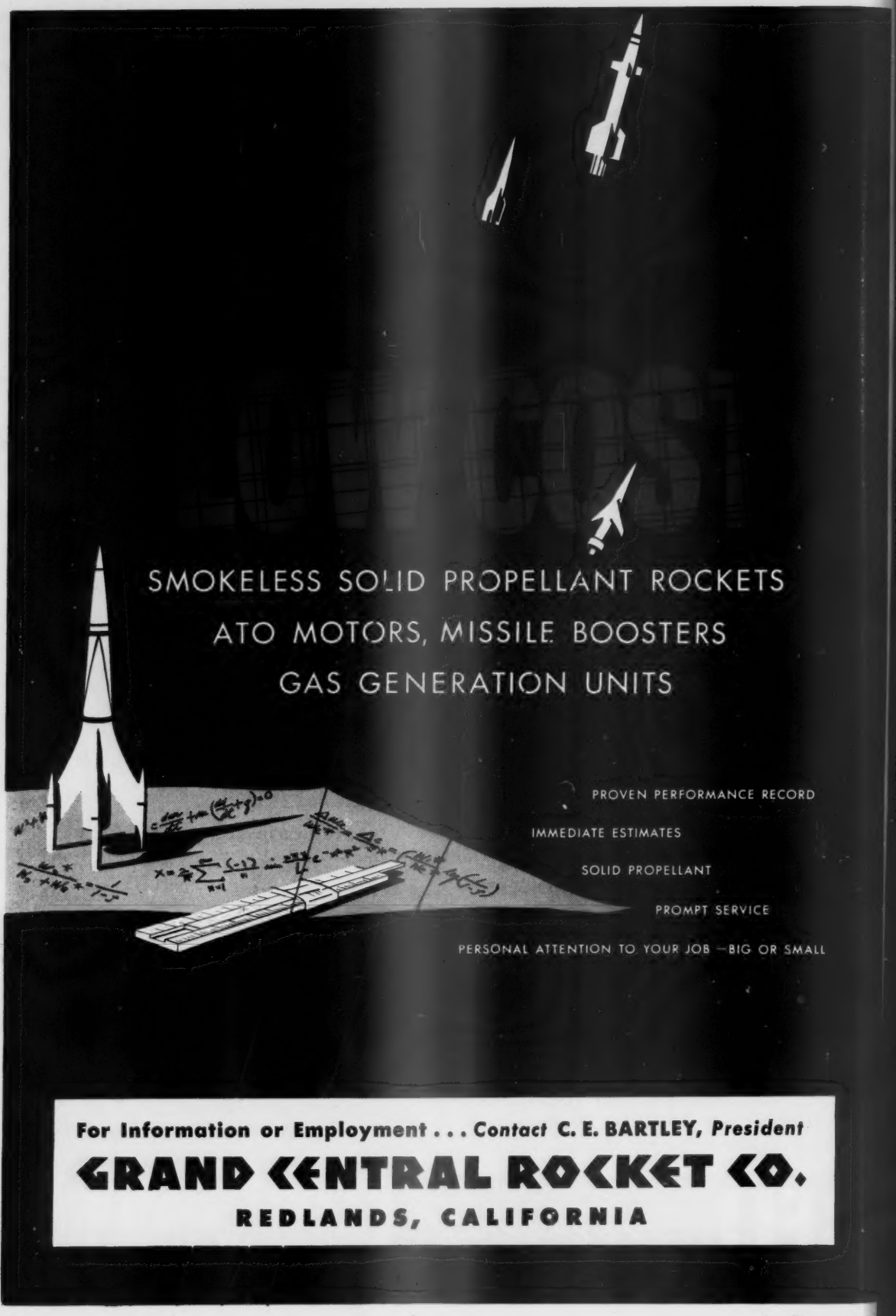
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